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PROJECT STINGRAE

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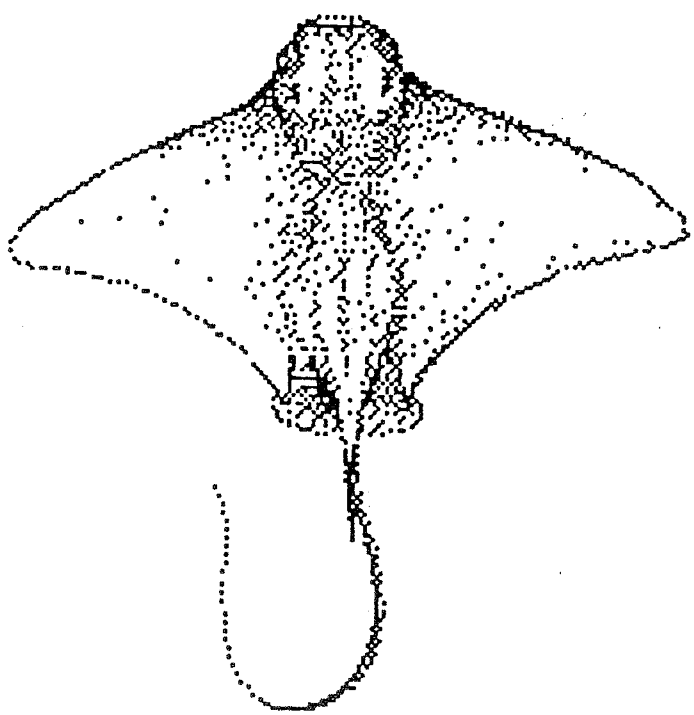


Table of Contents

Prelimary Design Considerations:

The study conducted under the project name STINGRAE (for Space Transportation Integrated Resupply And Automated Evacuation System) was designed as system intended to fill the need to rescue and supply the space station with an adequate support for performing missions envisioned for the year 2000 and beyond.

Because the number and type of STINGRAE missions perform in the specified time period would have a great effect on the configuration and effort was made to determine what the demand would be for the various types of subsystems visualized as within the scope of project STINGRAE. Each subsystem has specifications that must be accomplished. Seven categories of specific subsystems were analyzed:

1. Structures
2. Communication and Command Data Systems
3. Attitude and Articulation Control
4. Life Support and Crew Systems
5. Power and Propulsion
6. Reentry and Recovery Systems
7. Mission Managament, Planning and Costing

Specific structure requirements include: Placement of components to meet conflicting requirements, mass/inertia configurations, verify launch vehicle compatibility, drawings of layout.

Communication and Command Data Systems requirements include: Data rate estimates, antenna sizing/placement, geometry for antenna pointing throughout mission, rendezvous and docking, interations with the other subsystems.

Attitude and Articulation Control requirments include: Delta-V required for minimum maneuver scheme, attitude control modes, selecton and placement of AACS sensors, scanning and pointing requirements

implementation, fuel requirements/sizing, payload loading and unloading and interaction with other subsystems.

Life support and crew subsystems requirements include: Crew size vs. life support requirements, tank sizing, crew volume reqm'ts, threats (reasons for leaving space station) and interaction with other subsystems.

Power and propulsion requirements include: Power estimates, selection of batteries, solar cells, fuel selection/tank sizing, thrusters selection/configuration and interaction with other subsystems.

Reentry and recovery include: Size/shape, placement of components, dynamic and control, crew g forces, recovery method and interaction with other subsystems.

Mission management and costing include: Mission delta-v required, orbit insertion altitude and velocity, mission timeline and mission planning effect on subsystems.

These requirements served as the basis for the formulation of the STINGRAE spacecraft design.

STRUCTURES

Requirements

The main requirement for the structures subsystem in the request for proposal (RFP) submitted to group 3 is to design a vehicle structure capable of carrying supplies to and from Space Station Freedom repeatedly and bringing back humans (in an emergency) and waste to earth. While it is hoped that humans will not need to use the vehicle as a means of evacuation it must never the less make provisions for them.

To satisfy this, more specific requirements appear. For example, the vehicle must be capable of withstanding pressurization, it must protect itself against hazards encountered in launch, orbit and reentry, such as extreme thermal and structural loads. It must be reusable and safe and use tested reliable equipment.

STINGRAE is the response to this request.

General Description of STINGRAE

The Space Station Integrated Resupply and Evacuation System (STINGRAE) is shown in figure 1. It consists of an inside wall, a support structure, an outer micrometeorite shield covered in reusable surface insulation, vertical stabilizers[1], a body flap[2], a docking hatch[3], landing gear, and a small wing structure[4]. The overall length is 17 m, the width is almost 5 m, and the height of the vehicle is approximately 3 m. STINGRAE is constructed mainly

of conventional aluminum and covered in reusable surface insulation (RSI).

<u>Pressure Vessel Design</u>

The decision to pressurize the entire craft arose mainly from the logistics requirements for the vehicle. Approximate ratios of 2:7 for unpressurized mass : total mass and 1:2 for unpressurized volume: total volume made pressurization of the whole vehicle seem the most practical. The advantages of having a smaller pressurized area and a separate unpresssurized area were negated by the difficulties that arose regarding the distribution of space and therefore the construction of the vehicle to such a changeable factor. Since it was determined that all items in the projected payload would easily fit through the Space Station Freedom's hatch, it was decided that the entire cargo of the vehicle would be unloaded through that hatch and distributed through the space station's facilities.

The calculations for STINGRAES pressure vessel interior contain some assumptions they are as follows:

1. assumed cylindrical pressure vessel shape with a diameter equal to the widest part of the vehicle (This is over designing, but for lack of a more complex analysis this choice was felt to be prudent.)

2. used a yield strength of 2.89 (10^8) N/m^2 for aluminum 2024-T3. This value varies with the temperature of the material of the material and drops off rapidly for temperatures over 450 K, but

the thermal protection system (TPS) will assure that this temperature is not exceeded even during reentry heating.

3. assumed a safety factor of 2.5. Given the completely reliable and tested nature of the material used and the overdesigning mentioned in part 1 this was considered to be sufficient.

Using the equation below it is possible to calculate the pressure vessel thickness for the given conditions:

$$Y.S/(s.f.) = p(ri + t/2)/t$$

where:

Y.S = the yield strength of the aluminum (=2.89 (10^8) N/m^2)

s.f. = safety factor = 2.5

ri = radius of pressure vessel = 4.57m

p = is the pressure designed for (=1.013$(10^5)N/m^2$)

The thickness of the pressure vessel wall was found to be 0.2001.

Micrometeorite Shielding

Due to the length of time each vehicle will spend in space a major concern is insuring the structural integrity of the spacecraft during micrometeorite impacting. The micrometeorite shielding must be as thin and light as possible while still guaranteeing the pressure vessel will not be penetrated and spalling is minimal. The main considerations for the design of a micrometeorite shield are the diameter,mass, and velocity of the mircometeorites to be

expected, the material properties of the inner and outer walls of the vehicle, and the spacing between these walls.

Designing a single-wall spacecraft for a high probability of no perforations for a large area over a long time would necessitate an unacceptably large mass and multiwall systems have been shown to be less efficient than dual walls. It has been found through experimentation that the optimum design of walls for micrometeorite protection can be predicted with the following equation:

$$V = 12.566\ (1/E)(Str)(C)[(1-v)/(3+3v)]^{.5}\ (pd/m)^{2} S^{2}\ (ti)(to)$$

where: V=velocity of micrometeorite (km/sec) (avg. V=25 km/s)

m= mass of micrometeorite (gm) (=.0178 gm)

d= diameter of micrometeorite (cm) (= 1 cm)

v= Poisson's ratio of sheet material (=.33)

p= density of sheet material (gm/cm^3) (=2.77 gm/cm^3)

Str=critical stress of sheet (psi) (=42,000 psi)

E= Young's Modulus (psi) (=10.6(10^6) psi)

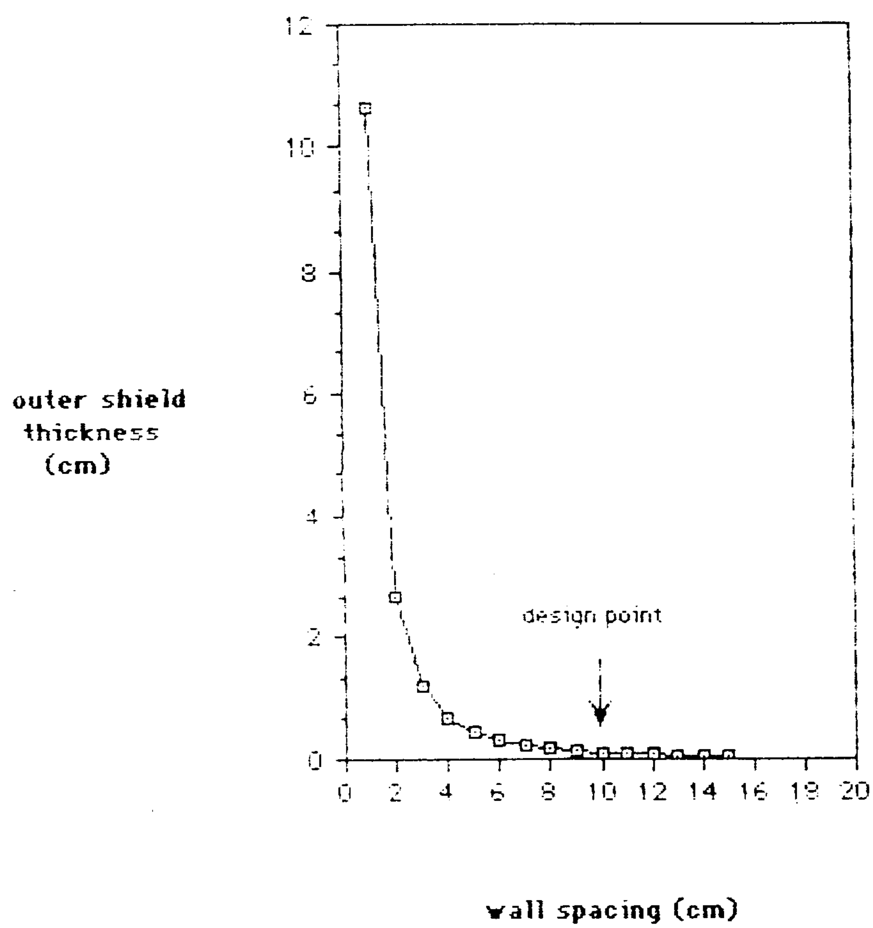
C= velocity of sound in sheet (km/s) (=5.140 km/s)

ti= thickness of pressure vessel (cm) (=.2001 cm)

S= sheet spacing (cm)

to= thickness of outer shield (cm)

OUTER SHIELD THICKNESS VS. WALL SPACING



Design selection for an inner wall of thickness ti=.2001cm is:
to=.1060 cm at a spacing of S=10 cm.

Figure 2

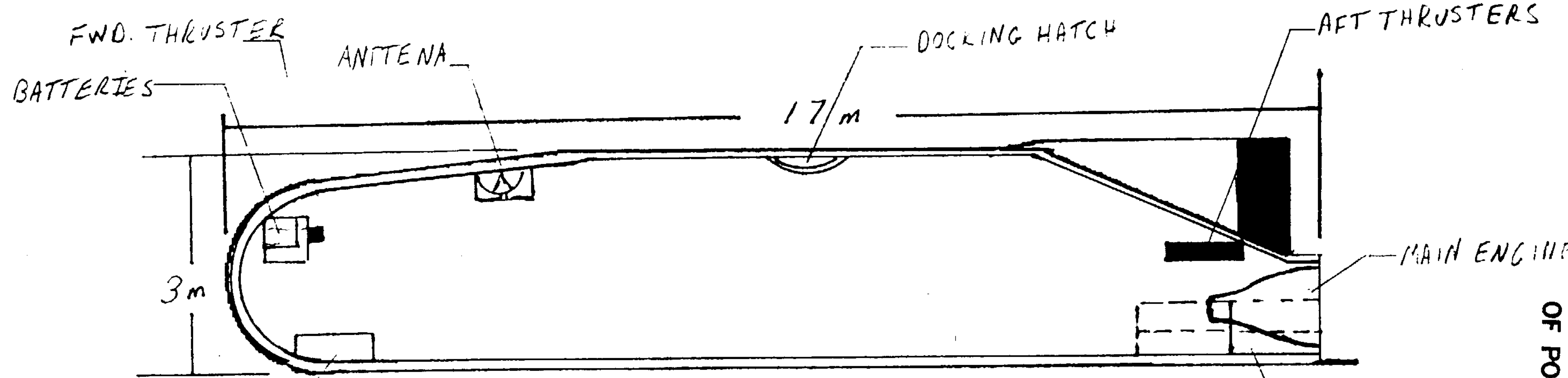
Figure 2 shows a comparison of sheet spacing vs. thickness of outer wall. The desired design minimizes both the spacing between the walls and and thickness of the outer wall (and therefore the mass). The design value is a spacing of 10 cm and an outershield thickness of .1065 cm.

Vertical Stabilizers and Body Flap

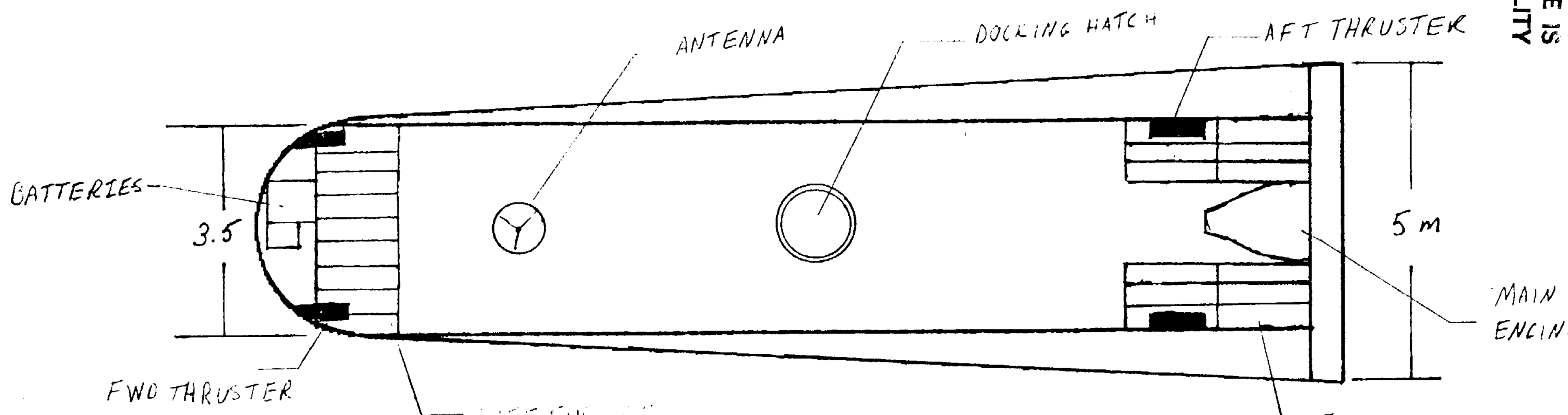
The vertical stabilizers on the back of the vehicle, each consisting of a structural fin surface, a rudder/speed brake assembly, a tip, and a lower trailing edge, are constructed of aluminum and covered with a thermal protection surface. The rudder splits vertically into two halves to serve as speed break during the landing phase. The back body flap, also constructed of aluminum, is designed to provide some thermal shielding for the back end of the vehicle during reentry and provides pitch control during the atmospheric flight phase following reentry.

Component Layout

The five subsystems having components to layout in STINGRAE are; power and propulsion, life support, command and data control, attitude and articulation, and reentry. On the following diagram the positions of the largest, heaviest items, having the most influence, are shown. The main objective in the positioning of components is to balance STINGRAE. The elements were laid out through the program INERT. This program takes into account the moments of inertia and centers of mass of each individual component and the outerhull of STINGRAE itself and calculates a center of mass and



SIDE VIEW



TOP VIEW

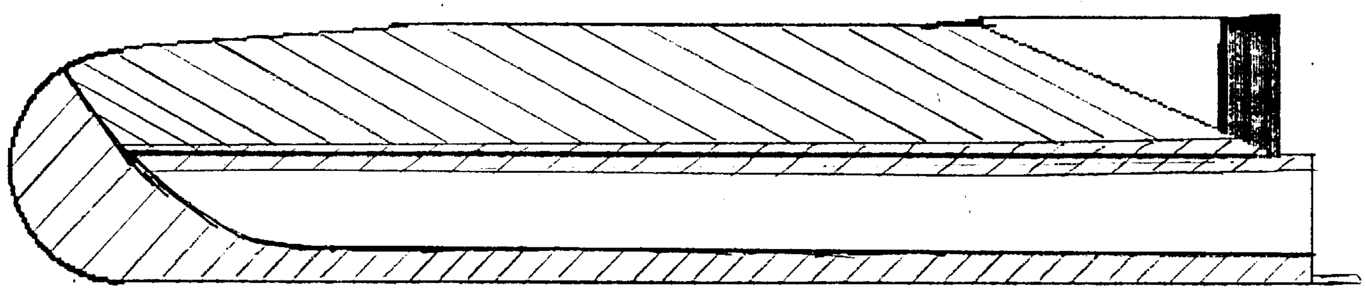
moment of inertia for the entire vehicle. The heaviest components have the most influence on the positioning of the center of mass; therefore they were used at opposite ends of the vehicle to balance each other out (i.e. the fuel tanks for propulsion are in the back while the life support tanks were kept in the front.). The variation in payloads make them impossible to specifically layout, so the optimal configuration for the vehicle puts the payload area as much in the center as possible.

Thermal Protection System

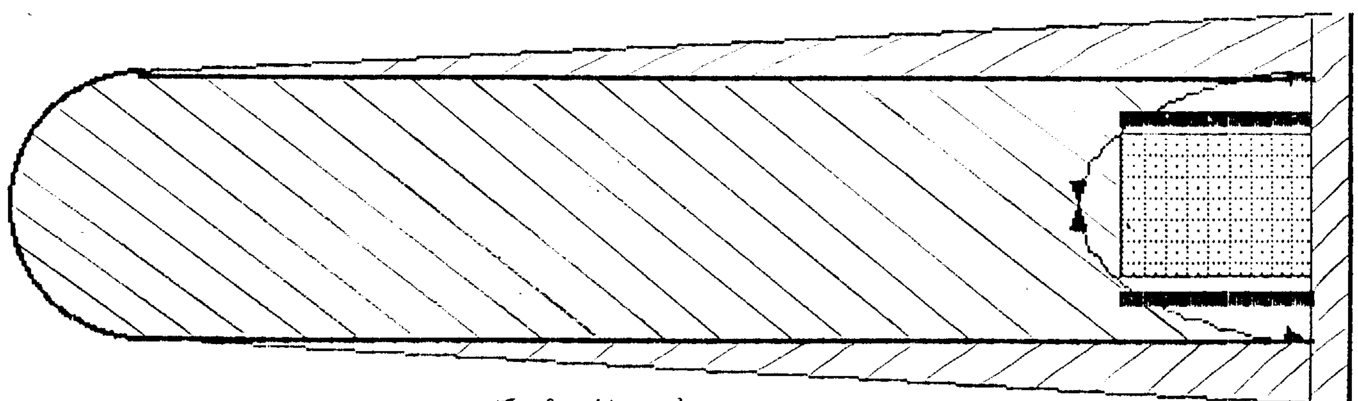
The thermal protection system (TPS) consists of the external heat shielding on the vehicle to protect the structure from excessive reentry heating. The optimal TPS minimizes the size, mass, complexity, and cost of the system, and maximizes ease of application, reliability, durability. To achieve this the TPS is composed of several different types of shielding, each one the optimum material for its temperature range. The minimum shielding must protect the primary structure to 450 K. The expected temperature of a craft is dependent on the outer mold line geometry and reentry velocity. For example sharp leading edges require the highest temperature shielding and the smooth upper surface can accept the lowest temperature shielding. Below is a diagram of STINGRAE ; the shaded areas represent the minimum type of shielding the ship will require for expected (approximate) surface temperatures. These three types of shielding have been studied as alternatives to the system used by the space shuttle.

TPS DISTRIBUTION

	Adv. Carbon-carbon
	BIMETALLIC
	TITANIUM



SIDE VIEW



TOP VIEW

The titanium multiwall panel (figure 4) , for up to 811K, is constructed of alternating layers of flat sheets of foil-gage titanium and dimpled foil gage sheets, diffusion-bonded to produce an integral prepacked tile complete with attachments.

The prepackaged superalloy bimetallic sandwich (figure 5), for up to 1255K, consists of fibrous insulation encapsulated by inner and outer panels, which are connected by a foil gage beaded sidewall.

The advanced carbon-carbon (ACC) standoff panel (figure 6), for areas above 1255K, is orthogonally reinforced with carbon-carbon ribbing and stands off the skin of the vehicle on posts. The effect is to prevent the buildup of excessive thermal stresses and strains.

Although these materials provide a considerable weight savings over the materials used in the space shuttle program and are therefore quite an improvement, it should be possible with further research to improve even these materials substantially.

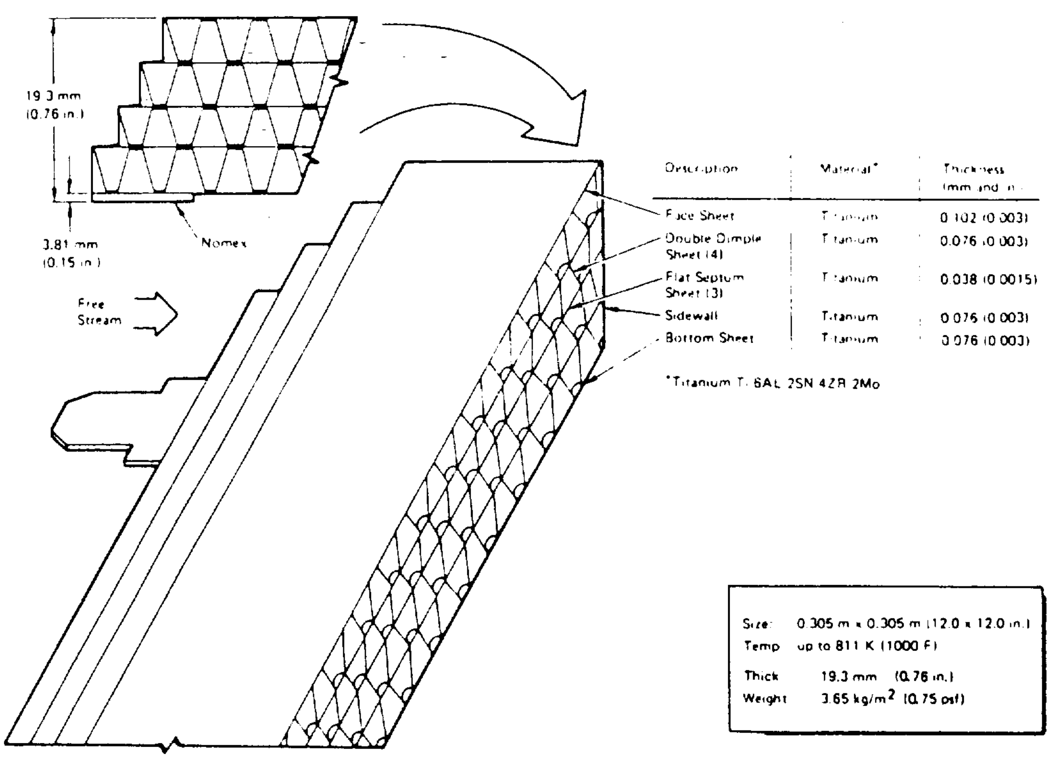
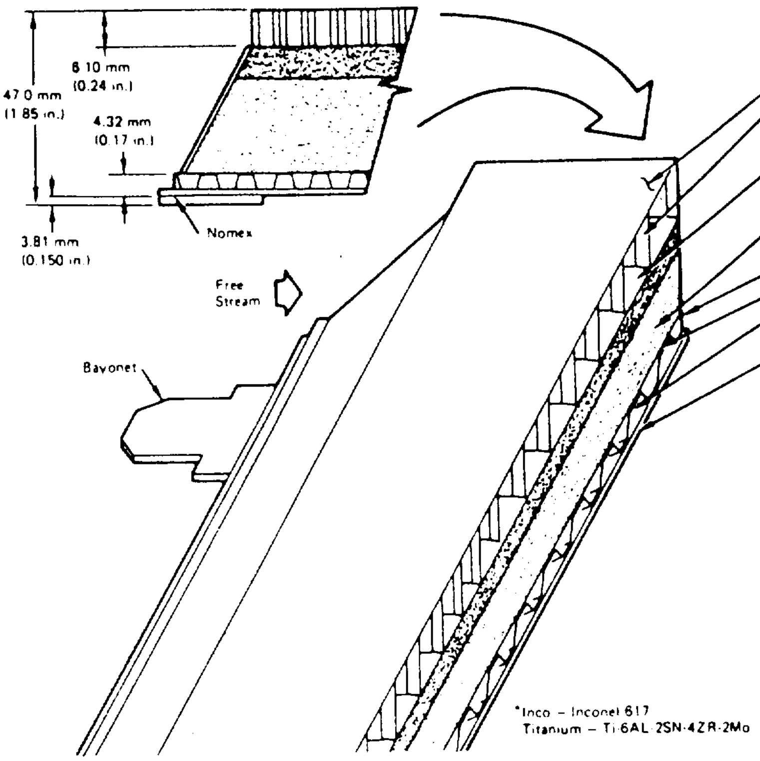


figure 4 (ref. 3)

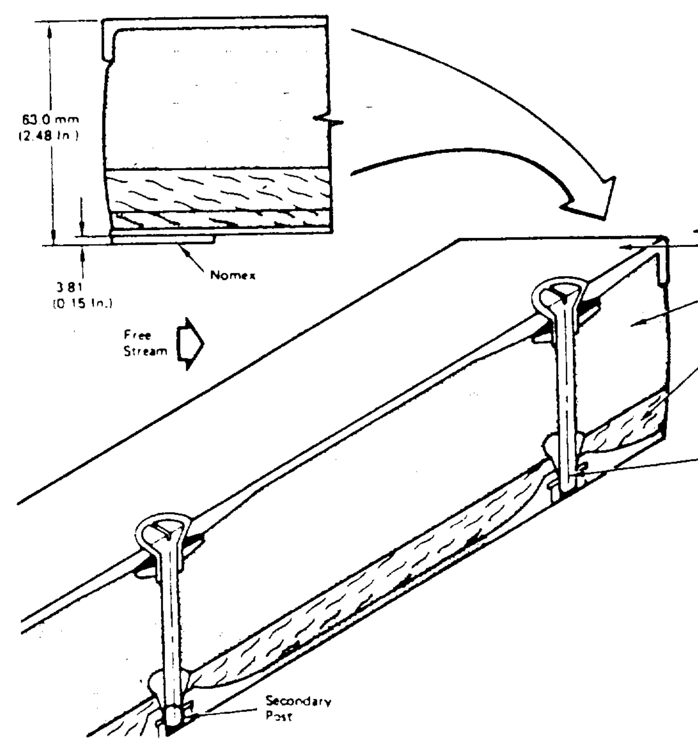


Description	Material*	Dimension (mm and in.)	
Face Sheet	Inconel	0.203 thk	(0.008)
Honeycomb Core	Inconel	4.78 cell 0.051 wall 5.84 thk	(0.188) (0.002) (0.23)
Inner Sheet	Inconel	0.076 thk	(0.003)
Insulation Batting	Cerachrome	12.7 thk	(0.500)
Insulation Batting	Q-Fiber	20.1 thk	(0.79)
Sidewall	Inconel	0.076 typ	(0.003)
Top Sheet	Titanium	0.076 thk	(0.003)
Double Dimple Sheet	Titanium	0.076 thk	(0.003)
Bottom Sheet	Titanium	0.076 thk	(0.003)

Size	0.30 m × 0.30 m (12.0 × 12.0 in.)
Temp	811° to 1366 K (1000° to 2000°F)
Thick	47.0 mm (1.35 in.)
Weight	7.31 kg/m² (1.50 psf)

*Inco – Inconel 617
Titanium – Ti-6AL-2SN-4ZR-2Mo

figure 5 (ref. 3)



Description	Material	Dimension (mm and In.)
ACC Panel	Coated ACC	Typ 1.78 (0.070) (6 ply)
Insulation	Saffil Alumina (1.74 Kg/m³)	27.8 (1.09)
Insulation Package	Q-Fiber Felt (1.74 Kg/m³)	25.6 (1.01)
	Astro Quartz (0.5 Kg/m³)	3.05 (0.120)
14 Uniposts	Columbium	2.75x 44.4 (0.187 × 1.75)
3 Secondary Post	Columbium	4.75x 44.4 (0.187 × 1.75)

1) YD 40 Ni Cr Al Support Posts
Used For Applications Below 1366 K (2000°F)

Size	0.91 × 0.91 m
Temp	- 0.91 × 0.91 m (36 × 36 In.)
Thickness	1,366 K to 1,755 K (2,000° - 2,700°F)
Weight	11.3 Kg/m² (2.31 psf)

figure 6 (ref. 3)

Thermal Control Subsystem

The thermal control subsystem (TCS) consists of the equipment required to maintain thermal control of all areas inside of the spacecraft outershield. This control should apply during all mission phases; including launch, earth orbit, space station docking and reentry. The TCS must be capable of:

1. radiating the excess internal heat generated by crew presence and onboard systems operations.
2. shielding the spacecraft's inner systems from external heating due to reentry, solar flux, albedo flux, and earth thermal radiation.
3. maintaining a "shirtsleeves" environment inside the spacecraft during periods when craft is in shade.

The thermal load relationship is:

$$Q_{sol} + Q_{alb} + Q_{earth} + Q_{int} = Q_{rad}$$

where,

Q_{sol} = heating due to solar flux

Q_{alb} = heating due to earth reflected solar radiation

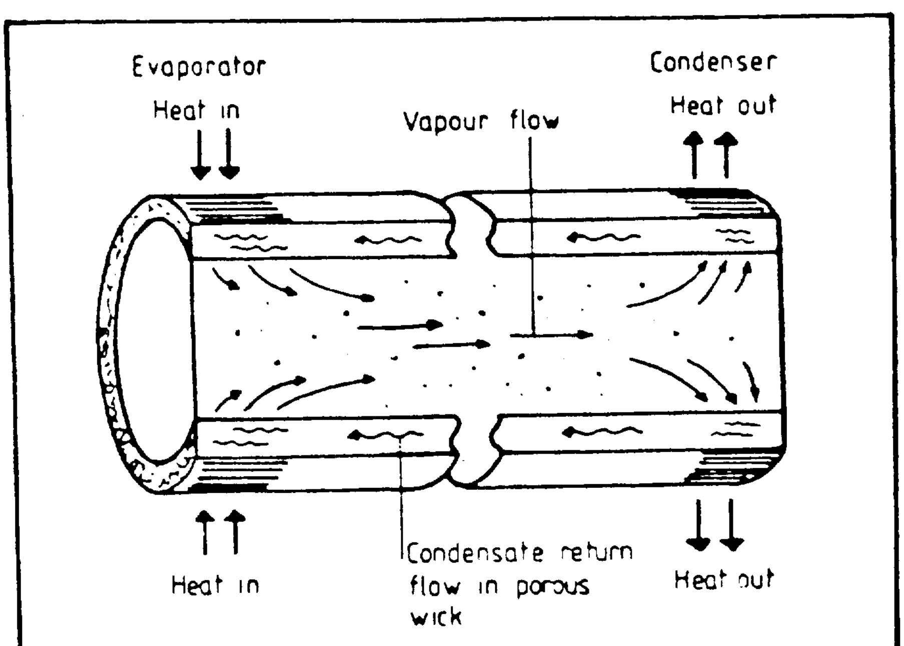
Q_{earth} = heating due to earth thermal radiation

Q_{int} = heating due to internal spacecraft systems

Q_{rad} = heat loss due to radiation

The outside structure of STINGRAE will be painted black on the leading edges and bottom for maximum radiation during reentry, and white on the rest of the surface to reflect the majority of solar radiation.

The amount of heat generated inside STINGRAE will vary according to number of crew members, activity of systems, and length of time spacecraft is occupied and active. This transient heating will be controlled in part by by the presence of a thermal capacitor (TC). The TC will be looped through a heat pipe system circulating throughout the ship. This heat pipe system will transfer heat from warmer to cooler regions of the ship by means evaporation and condensation of ammonia in aluminum pipes. The primary function of the TC is to assist in providing a steady-state thermal environment for the spacecraft by alternately acting as a heat source or sink. During times of excessive internal heating the TC will absorb much of the heat in the loop and return it during cooler periods.



HEAT PIPE (ref. 5)

In the event that the internal thermal loads exceed the capabilities of the heat pipe/TC combination the system will be linked to a radiator panel located on the sloping back face of the vehicle between the vertical stabilizers. When not in use the main panel will be covered by another panel with reentry shielding on the outside and a radiative surface on the inside. This outer panel will be hinged at the top and swing out to a vertical position thereby increasing the surface area of the radiator by a factor of two. These inner panels will be shaded from solar flux, albedo flux, and earth thermal radiation by the vertical stabilizers on either side and the back of the outer panel itself.

Reentry heating is the highest thermal load the craft will be expected to experience. The thermal protection system is capable of shielding the outer hull of the spacecraft to about 450K (above this temperature the yield strength of aluminum drops rapidly). To keep the environment of the spacecraft from overheating due to this temperature the inner wall must be insulated. Customarily, multilayer insulation (MLI) also called the "thermal blanket" is used. It is made up of several layers, each acting as a low emmitance shield separated by low-conduction spaces., for example, layers of Mylar and Kaptan foil each almost .25mm thick aluminized on one side. A typical ten layer blanket with a total thickness of 5mm would be equivalent to 500mm of conventional insulation.

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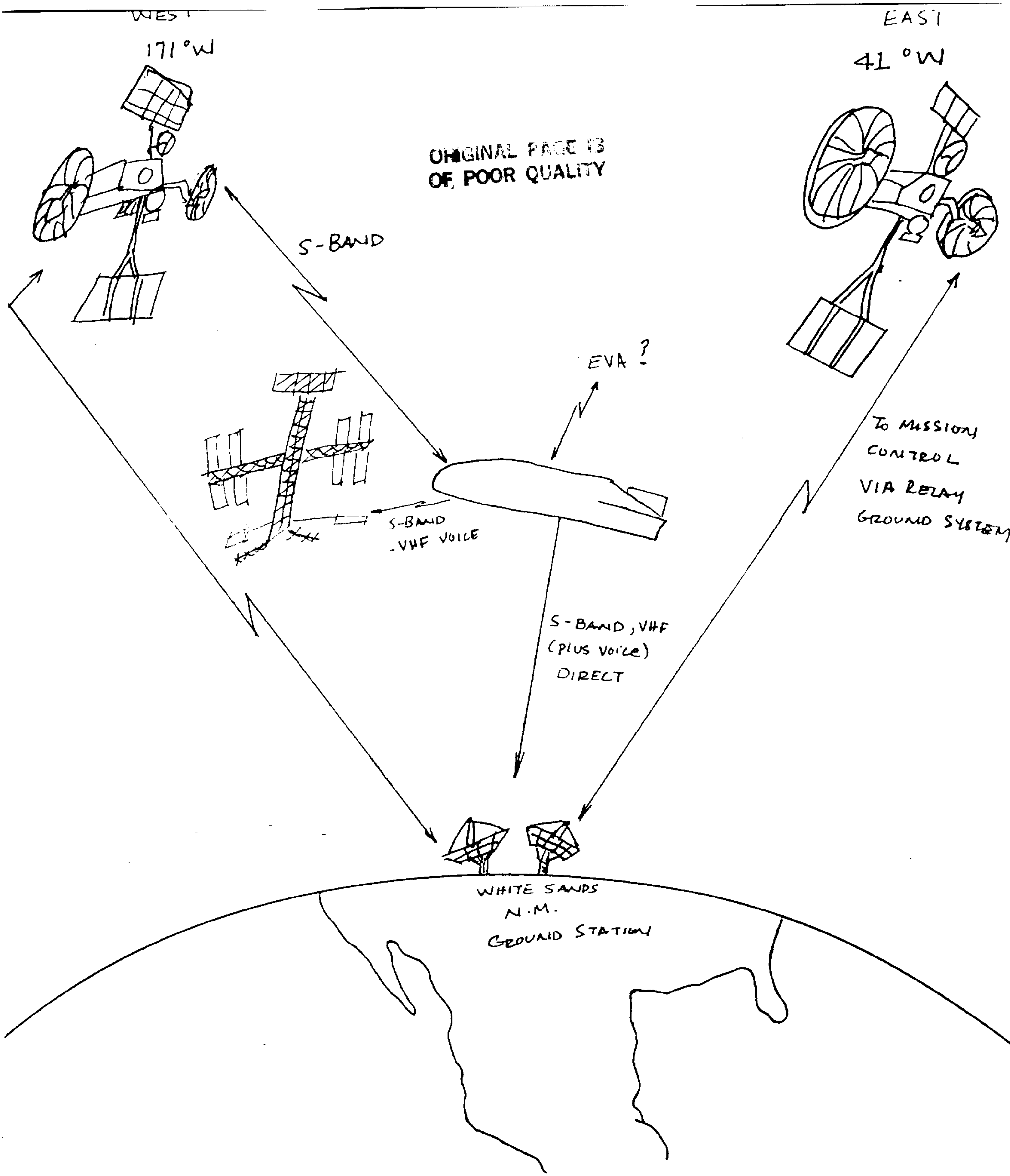
ON BOARD COMMUNICATION SYSTEMS

Design Considerations. Functional requirements for project STINGRAE communications system include collecting telemetry from the subsystems, sending telemetry to the ground , command power switching, sending commands to the subsystems and crew support avionics. The primary function is to transmit data back to the earth. The three basic forms of this data are: scientific, engineering (which includes spacecraft's health), and commands.

Some STINGRAE missions would require engineering and scientific information-gathering. It is necessary to obtain voluminous amounts of data on the condition of the spacecraft, astronauts or cargo, and the performance of the subsystem. In the design of the performance of the system telemetry will be sent to the ground. Automated docking with the space station will be controlled by an on board computer.

Considerations for design also include compatibility with the tracking and data relay satellite system (TDRSS). TDRSS consists of two communications relay satellites, TDRS-east and TDRS-west. These are positioned in the geosynchronous orbit approximately 41^{o} W and 171^{0} W longitude, respectively. The TDRSS spare is located at 83^{o} W longitude. The TDRSS relays signals between the ground station, *(in White Sands, New Mexico)*, and orbiting spacecraft and user control centers, below 12,000 km above the Earth.[1] Since the space station is located between 290 and 430 km, the STINGRAE should be compatible with TDRSS. Refer to *figure 1* , for STINGRAE's compatibility features with TDRSS.

In addition to compatibility with TDRSS the system must be standardized within itself. This standardization comes from the requirement for versatility due to the variety of missions whether it be



STINGRAE COMPATIBILITY WITH THE TDRSS (FIG. 1)[3]

transporting cargo or scientists. It was first required the STINGRAE would maneuver and rendezvous with orbiting platforms, but because of too high åDv requirements in the transfer of orbits, this required communication capability was dismissed. *(See propulsion and power for further details.)* Standardization with the system, however, makes different parts of the system serve as backups for each other making the system reliable.

Communication System Configuration and Design. A major question to be resolved in the design process is of which band or antenna configuration is optimal for STINGRAE's performance. Using data from the Apollo missions and the Space Shuttle's use of TDRSS, which most space communications of this day use, the best system for STINGRAE's requirements were chosen.

Like Apollo, STINGRAE will use a VHF Radio link for communicaiton and telemetry. For near Earth orbits this system can be used until the s-band system is applied. This system also provides a secondary back up. Although not a requirement, the VHF system could be used for a radio communication link with an extravehicular astronaut (EVA) with direct ground station links. The VHF system is used in conjunction with s-band , phase modulated (PM), frequency modulated (FM), radio links with ground stations. STINGRAE will have four quarter - wave monopole whip antennas located in different areas of the spacecraft and will be offset to provide near-omnidirectional coverage. *Figure 2* illustrates a standing wave on quarter - wave antenna.

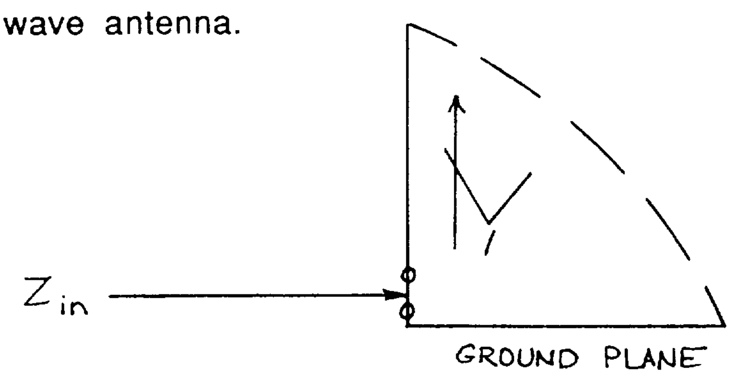


Figure 2: A standing wave on quarter - wave antenna.[3]

The spacecraft will use this VHF system in close range ground station passes. The VHF system has a 5 watt output and a frequency of 296.8 MHz.[2] This system also provides communications while landing. Landing communication frequencies need only be from 150 to 700 MHz, which appear to be a good compromise for inexpensive systems that do not need more accuracy than a nautical mile, (1.85 km). *(See Mission Managament and Planning for futher details on costing.)*

Since the Apollo, the s-band direct ground station link system has been upgraded. The s-band direct uplink provides 32- kilobit delta - modulated voice channels and a data (command) channel. The resulting uplink rate to STINGRAE is 72 kilobits per second.

The s-band direct to ground stations downlinks, 2, 32- kilobit digital voice channels with delta modulation. Downlink also provides 128 - kilobit telemetry, which results in a time-division-multiplexed data rate of 192 - kilobits per second. [2]

The Space Shuttle uses two separate radio frequency links though the tracking and data relay satellites. A s-band link with low - gain antennas can be used. (Low - gain causes a wider band width, therefore, this is omnidirectional.) When the power is increased a new k-band link with even greater capability than the s-band link can be used. For STINGRAE's purposes, however, the high power antenna, the k-band will not be necessary although, could be added if STINGRAE's capabilites ever needed to be extended. Like the space shuttle, the STINGRAE will use a low power s-band antenna which acts as an omnidirectional type antenna can be sent to TDRSS' 3.81meter, (12 $^1/_2$ ft.), s-band dish. This particular set up can be operated in a excess distance of 40,744 km or 22,000 nautical miles by using STINGRAE's .9 meter , (3 ft.), s-band antenna. The space shuttle ranges in transmitted power from 10 watts to 100 watts on the

low power, s-band system. The STINGRAE will transmit a maximum power to TDRSS of 100 watts. The s-band link antenna receives and transmits telemetry , voice and commands. The schematic in *figure 3* shows the distribution of data through STINGRAE's antenna components.[4]

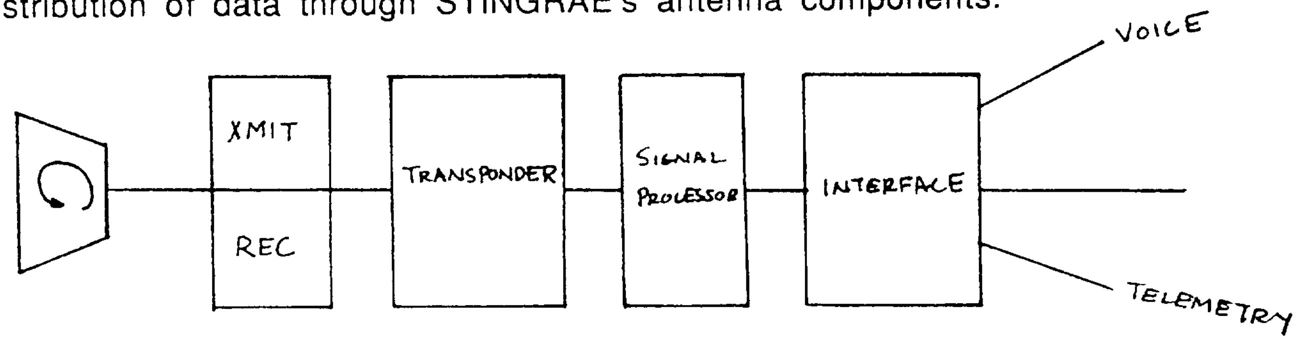
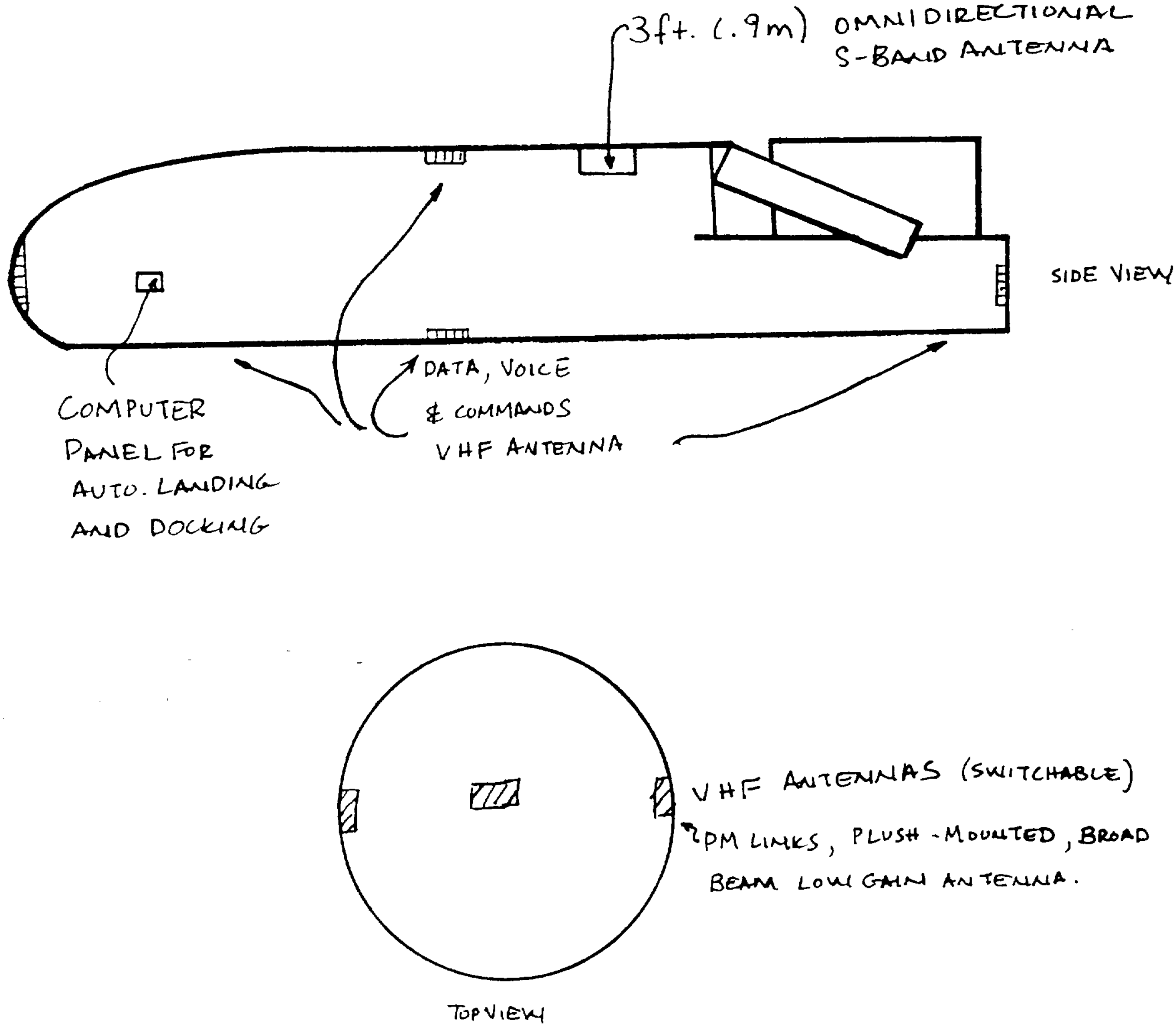


Figure 3 : Schematic of antenna distribution.[4]

The s-band forward link mode consists of one 24 - kilobit, delta modulation voice channel plus 8 kilobits of encoded communication data. The s-band return link consists of two, 32 - kilobit, delta modulated voice channels and 64 - kilobits of phase coded modulation telemetry. To show the entire component layout of STINGRAE's communication and control systems refer to figure 4.

STINGRAE antenna design required to cope with the effects of thermal protection system (TPS) tile, overlays the flush mounted antennas. This tile is subject to the wear and tear of repeated atmospheric reentries since each STINGRAE will fly many missions to and from the space station. (See structures for further detatils of TPS.)

The docking mechanism of STINGRAE will be compatible with the space station docking adapter. STINGRAE will use an optoelectronic docking system which uses light emitters, sensors and microcomputers to automatically control the approach of the spacecraft . The range of the automated docking is from the distance of about 1 km to and few centimeters.[5] *(See Attitde and Atriculation Control for details of controlling STINGRAE.)*



STINGRAE COMPONENT LAYOUT

FOR COMMUNICATION AND CONTROL (FIGURE 4)

Break Down of Communication Components

Components	Power	Volume	Weight
S-Band Antenna................	100w	.004m^3	6.612 km
4 VHF Flush Mount Antennas................................	20w	†	†
Signal Processor..................	45w	.012 m^3	37.468 km
Transponder........................	28w	.007 m^3	33.060 km
Automated Docking and Landing........................	20w	†	†
Computer panel..................	5w	†	52.896 km
TOTALS............................	250 w	.023 m3	130.036 km

† indicates data not found

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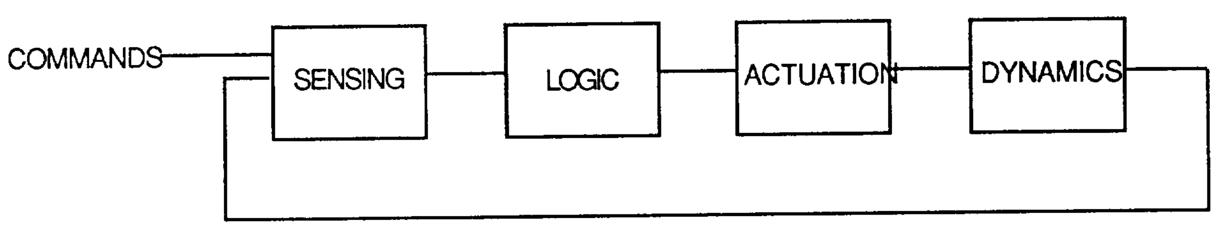
ATTITUDE AND ARTICULATION CONTROL OF STINGRAE

The STINGRAE's attitude control system has certain requirements. These are control of the spacecraft's attitude, pointing device control, and payload loading and unloading.

To control the attitude, STINGRAE's system will consist of four major functional sections. They are sensing, logic, actuation and vehicle dynamics. The sensing function determines the spacecraft attitude. The logic programs the electronic signals in a correct sequence to the torque producing elements, which, in turn, stabilize the spacecraft about its center of mass. The resulting motion is then sensed by the vehicle sensors which thus close the loop of the STINGRAE's attitude control system.[2]



The basic type of attitude control will be provided by the STINGRAE's three axis active control system. This system consists of two main classes. One is the mass expulsion, which is pure jet system and the other is a momentum exchange system, which consists of control moment gyros and pitch wheels. The STINGRAE will use an integrated version of both of these systems to compensate for the internal and external torques. An integrated

2

system thus will require a logic that is capable of coordinating the efforts of both of these systems and therefore this vehicle will demand the latest in computer science technology. External torques, mentioned above, arise through the interaction of a vehicle with its environment. Some type of external torques are gravitational, aerodynamic, meteorological impact, and radiation. Calculation of external torques requires a specification of both vehicle properties and of the space environment within which the vehicle is situated. Internal torques, on the other hand, are caused by fuel sloshing, control jets, and the motion of the crew.[1]

In addition to correcting for the above perturbations, an attitude control system will allow the spacecraft to be oriented or rotated on automatic command into a specific direction to permit the pointing of instruments and docking with the space station. These maneuvers will require very accurate application of small torques.

The STINGRAE's attitude has to be controlled about three mutually perpendicular axes, each with two degrees of freedom giving a total of six degrees of rotational freedom. In order to apply a true torque it is necessary to use two thrust chambers of exactly the same thrust and equal start and stop times, placed an equal distance from the center of mass. In order to get the maximum torque, the thrusters will be placed at maximum distance from the center of mass satisfying the equation $\mathbf{T}=\mathbf{R} \times \mathbf{F}$. Where the $\mathbf{T}$ is the torque produced, $\mathbf{R}$ is the distance from the center of mass and $\mathbf{F}$ is

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1

the force produced by the thruster. There is a minimum of twelve thrusters required in this system, but with STINGRAE's geometrical design, ten thrusters in front and ten in the back of the vehicle will be used. The placement of the thrusters is shown in figure (3).

Control torques in STINGRAE's active attitude control system will generally be obtained from a cold gas. The main reason for the use of cold gas is due to safety requirements. The cold gas system will use an inert gas of nitrogen stored in a high pressure vessel with initial pressure up to 400 atmospheres. The main reason the nitrogen was chosen was because it offered the best theoretical specific impulse vs. density ratio. This is illustrated on the graph in figure (1). The gas will be passed through one or more regulators so that the thrusters operate at nearly constant pressure. The thrust range will be between .01 to 5 lbs and will provide a specific impulse of 60 to 80 seconds.[7]

The maximum Delta-V required for STINGRAE in its flight was assumed to be .1 m/s and by using the equation **:**

$$\mathbf{P=W\ (\ exp\ (\ Delta\text{-}v\ (\ g\ x\ Isp\ )\ -1\ )}$$

Where the **P** is the propellant required, **W** is the weight of STINGRAE, and **g** the acceleration due to gravity. For thirty maneuvers and a safety factor of 1.5 the total propellant of nitrogen was estimated to be 130 kg. The propellant will be stored in four high pressured tanks and the placement of the tanks in the vehicle is in fig(3).

The other half of the active system will consist of the momentum exchange system. In this reaction wheels or control

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7

moment gyros could be used. The STINGRAE will use the control moment gyros because control moment gyros compared to reaction wheel offers more torque capability with lower power consumption, as well as lower weight and size for the same performance capability.

A cluster of three control moment gyros will be used to produce torque in pitch, roll and yaw axes. The reaction torque exerted by the control moment gyro rotor on the gimbal is :

$$\mathbf{T} = d\mathbf{H}/dt - \mathbf{W} \times \mathbf{H}$$

Where the **T** is the torque produced, **W** is the angular velocity of the control moment gyro and **H** is the total momentum. The amount of torque produced will be between .01 to 10^3 ft-lb.[2] A total attitude with control moment gyro system is shown in figure (2).

Attitude referance for the STIGRAE will not employ Euler or gimbal angles. The orintation of the spacecraft body to the referance coordinate system will be specified by a nine element direction cosine matrix. A four-element equivlent quaternion is extracted is used from this matrix and the flight control equations and coordinate transformations are formulated exclusively in terms of quaternions. The quaternion formalism was adopted for use because it offers computational efficiencies in terms of memory usage and execution time as well as a convenient physical interpretation of the spacecraft.[3]

Selection and placement of sensors:

2
3

During the STINGRAE's mission it will be necessary to determine the vehicle's attitude relative to an inertial frame of reference. The two type of sensors chosen for this are the rate sensors and attitude sensors. Looking at the attitude sensors the STINGRAE will contain the star tracker. The star tracker chosen is the Bal Aerospace Systems Divisions' Standard Star Tracker. It is chosen because it offered versatility, high sensitivity and flight proven design. The tracker incorporates all the landmark features, plus the convenience of a self contained power converter, digital position outputs, and several performance options which increase its utility. Its combination of large field of view and high sensitivity enable it to detect and track stars in any portion of the sky, thereby placing no constraints on spacecraft orientation. This tracker is equally useful for closed loop attitude control or star field mapping for precise attitude determination.[1] The placement of the tracker is shown on figure (3).

Another type of attitude sensor on the STINGRAE will be the sun sensor. This sensor will be used for backup in case of failure of star sensor. The specific type chosen is the Digital Sun Sensor. This sensor produces a digitally coded output that can be used directly by the attitude determining electronics. This sensor uses a number of solar cells arranged in a digital code form. This sensor has given high sensitivity and a field of view ranging from several arc-minutes to 128 by 128 degrees and resolution of less than an arc-seconds to several degrees.[1]

1

The rate sensors on the STINGRAE are made of fiber optic gyros. These gyros are still in research stage but before 1994 these gyros will be able to perform the same sensing tasks as the traditional mass gyros and the laser gyros available in the market today. The main reason this type of gyros is chosen over its competition is that it offers some great advantages. These advantages are its small size, ruggedness and the prospect of modest cost. As a "strapdown" device it does not require expensive gimbaled mounting system and it is free of low-rotation-rate-lock in that causes other gyro types to produce false zero outputs.[6]

Accelerometers. During the ascent portion of the space vehicle's flight, it will be subjected to large forces caused by the thrust of the propulsion system and by aerodynamics lift. These forces must be measured to provide guidance information and keep the maneuvers of the vehicle within safe limits. The accelerometer is a device which is capable of measuring these forces applied to it. Since it is necessary to know the forces acting along all three axes of the spacecraft, three accelerometers mounted along orthogonal axes will be used. The type used will be the quartz resonant accelerometers. It employs a proof mass suspended from dual double tuning-fork, fabricated on a quartz substrate using metal film deposition techniques. This yields a design whose performance is relatively unchanged by environmental effects.[6]

The payload loading and unloading in the STINGRAE basically will be done manually. All the payload taken up will be able to fit

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6

6

through the docking adapter hatch. There is an assumption that there is a lift arm attached to the space station and for heavy objects this arm maybe used.

In summary, the STINGRAE spacecraft will be attitude-stabilized by a three axis active attitude control system utilizing an integrated on-off jet actuators and momentum exchange of control moment gyros. The sensing units of gyros and trackers will give a sensing rate of internal and external torques and will provide other necessary attitude data. The total system is shown in figure (4).

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Data from "Untitled Data #1"

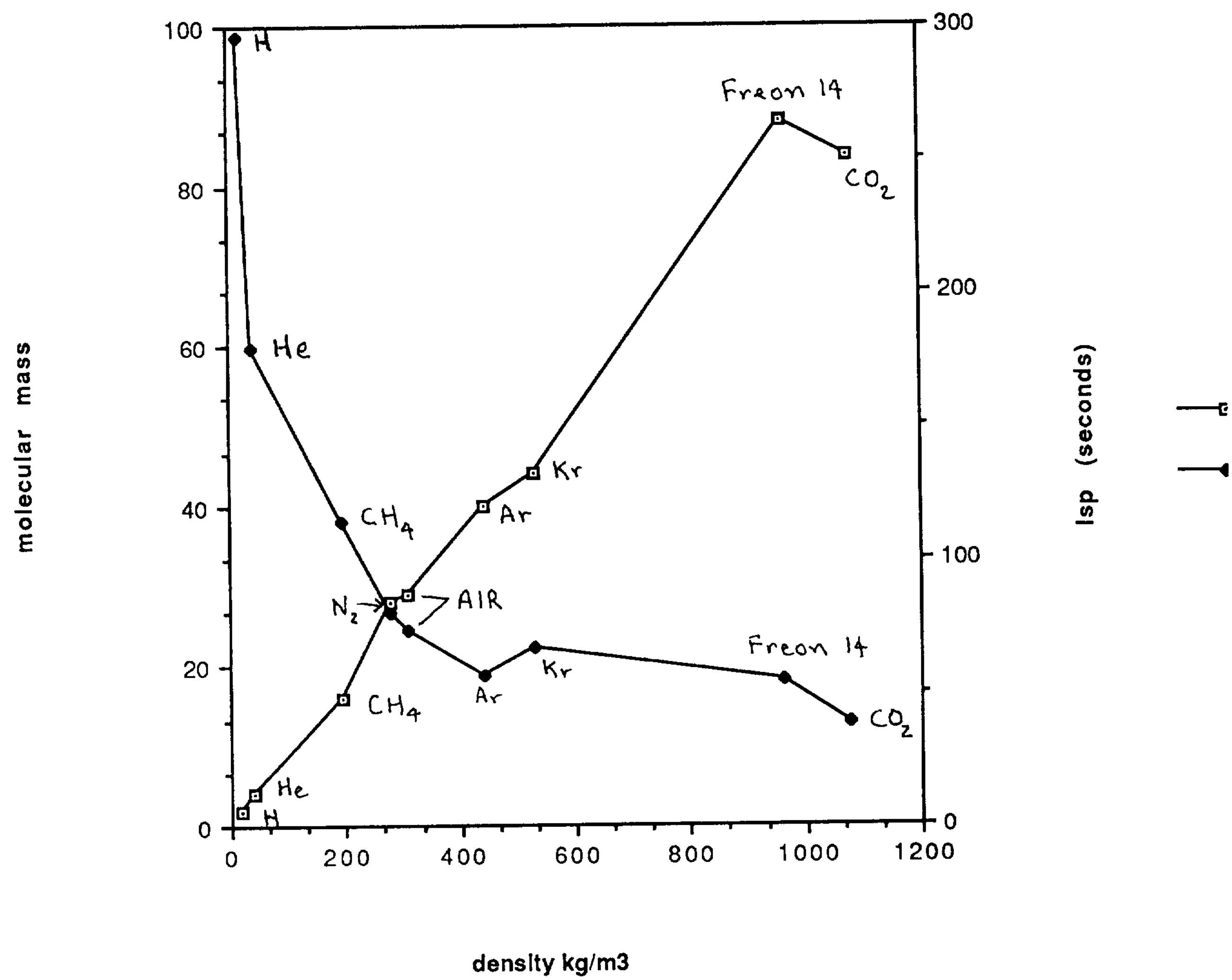


Fig 1

Reference - 7

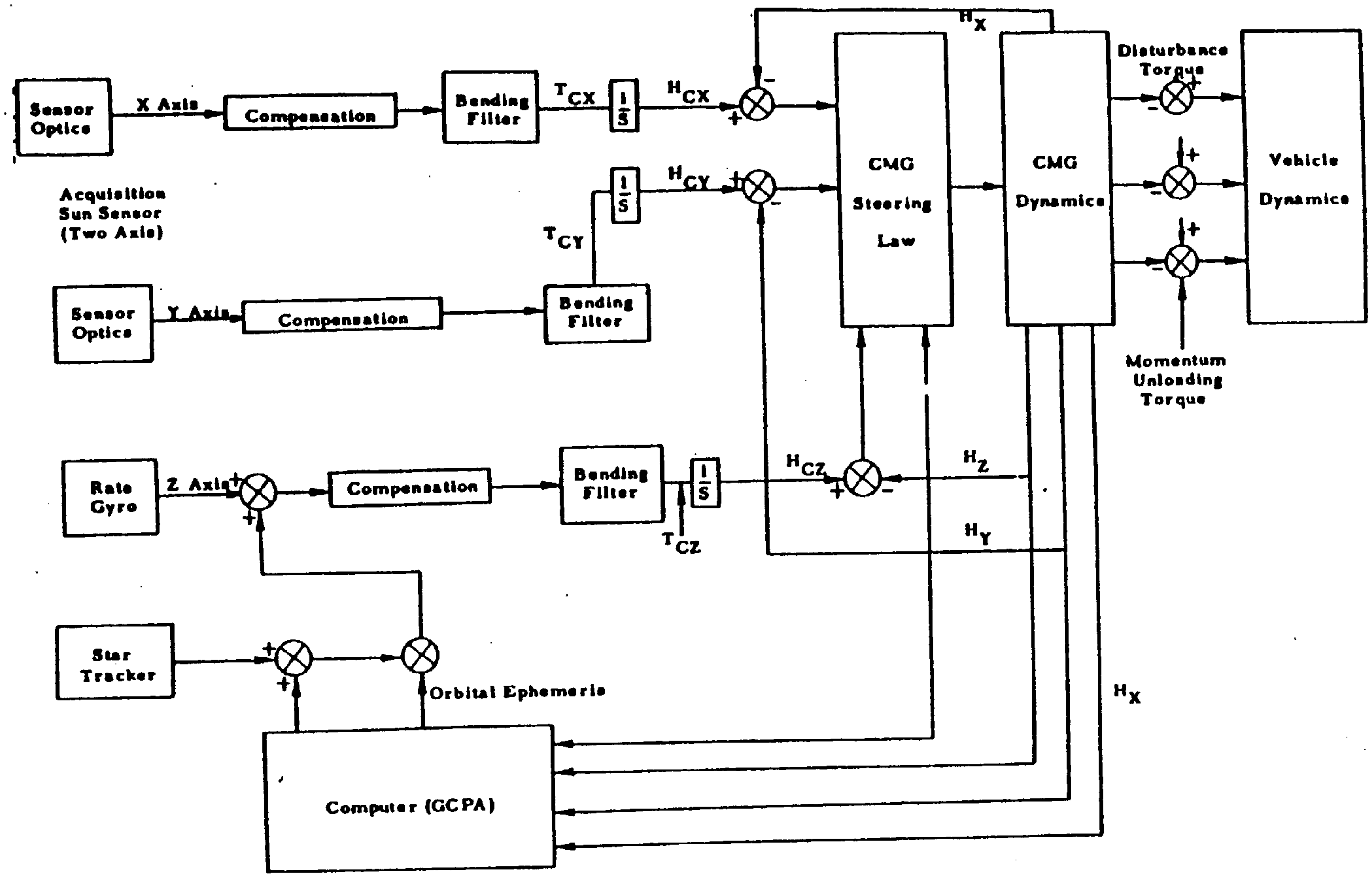
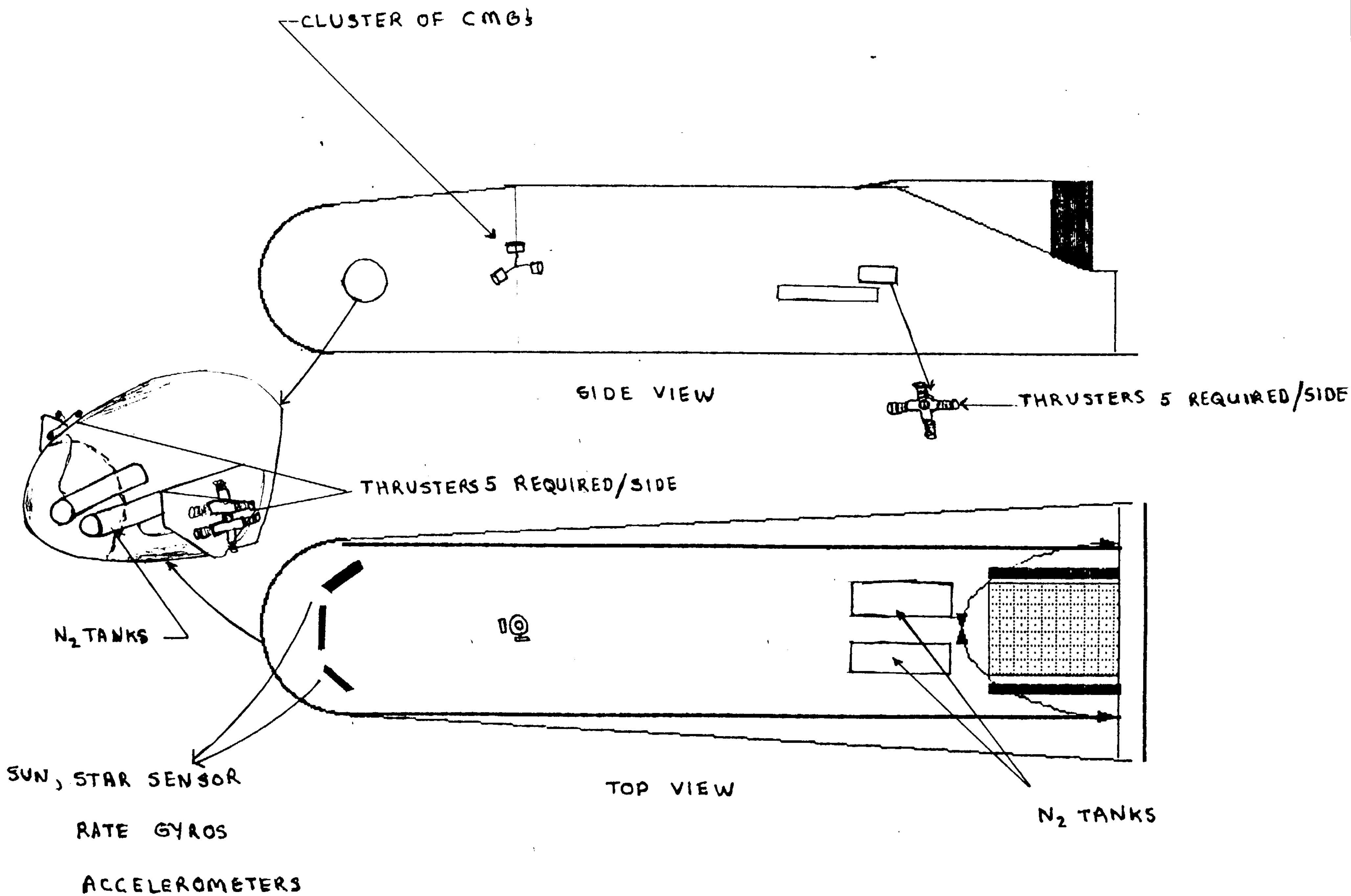


Figure 5.18. Simplified CMG Control Subsystem

Ref - 2

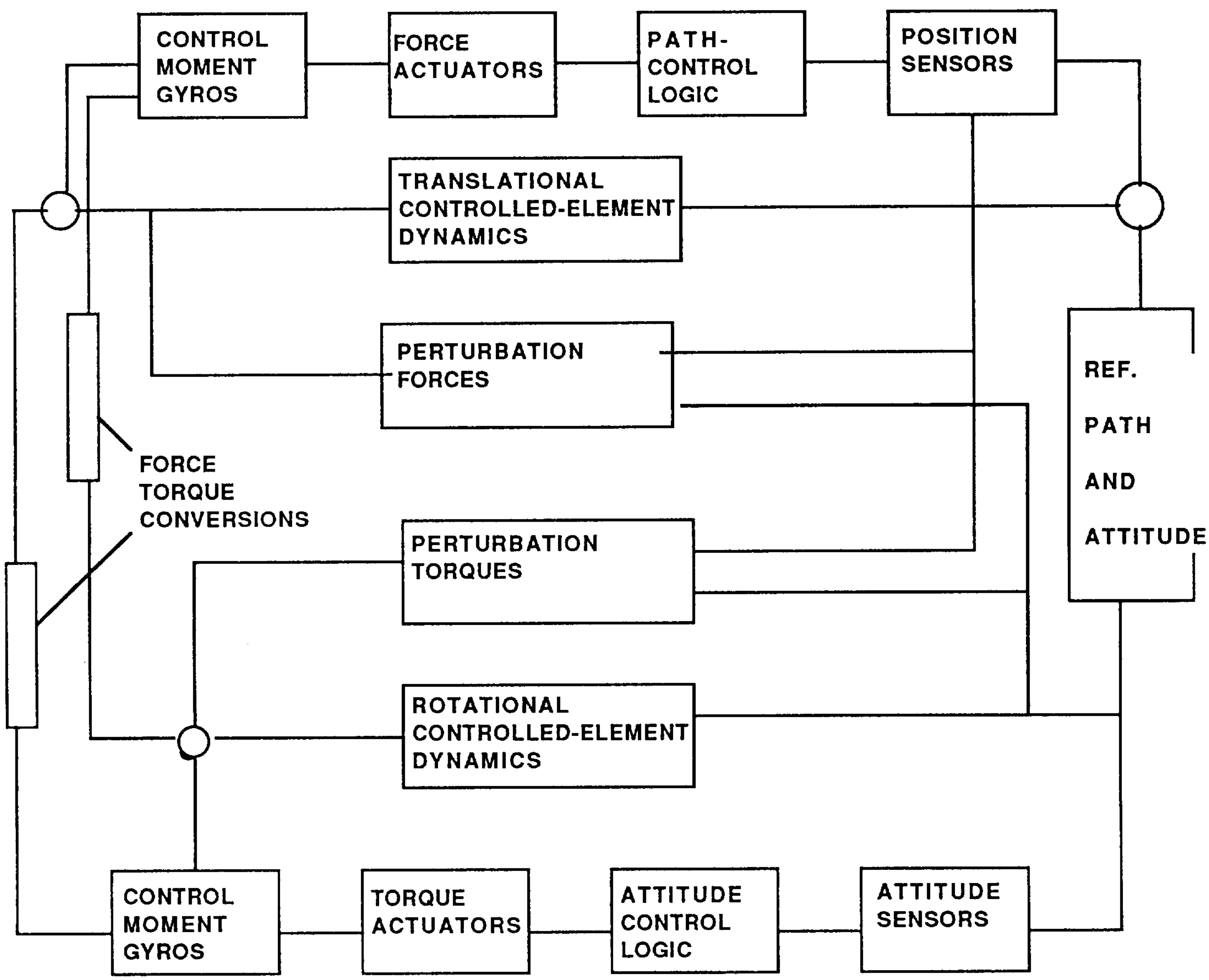
Fig - 2

Fig-3



CLUSTER OF CMG's
SIDE VIEW
THRUSTERS 5 REQUIRED/SIDE
THRUSTERS 5 REQUIRED/SIDE
N2 TANKS
TOP VIEW
N2 TANKS
SUN, STAR SENSOR
RATE GYROS
ACCELEROMETERS

Stingrae Total Attitude Control System



REFERENCE 2

Fig 4

Power

The power system of STINGRAE is required, by the RFP, to meet certain specific and derived requirements which are: to meet all subsystem power request and to do so with a system protected against single failure destruction, to identify levels of power consumption throughout the mission including peak consumption and space station power taxation, and to be low cost, simple, and light weight.

In response to these constraints, the power system of STINGRAE is as follows. The power system consists of four source components which perform five individual operations, each of which is dependent upon mission time. The mission divisions are as follows: launch to separation from Titan IV, separation from Titan IV through rendezvous with Space Station Freedom, attachment with Freedom, separation from Freedom to final orbit insertion, reentry through final taxi. Storage batteries provide two of the four power sources while the other two sources are externally provided, the Titan IV and space station Freedom.

Just before launch, the entire power system will become independent of ground supply and from this point until just before separation, the Titan IV will supply "stand-by" power to the attitude control system and full power to the life-support system of STINGRAE (see figure 20). Seconds before separation the primary

power system will become operable and fully activate the attitude control system.

The primary source of power originates from a collection of Silver - Zinc (Ag - Zn) cells. These cells form the main battery system which supplies the power from Titan IV separation through rendezvous with Freedom. This main battery system, after recharging at Freedom, also supplies the power from space station separation to final orbit insertion. The system will deliver a maximum power of two kilowatts per hour for sixteen hour at a depth of discharge of eighty percent. Since this time interval will far surpass all estimates on elapsed time from station separation to landing, it therefore will serve as a safety buffer. In the event of a station separation without a reentry, i.e. an emergency evacuation and later return to station, it is possible to maintain two kilowatts per hour of power for twenty four hours but this will require the batteries to completely discharged.

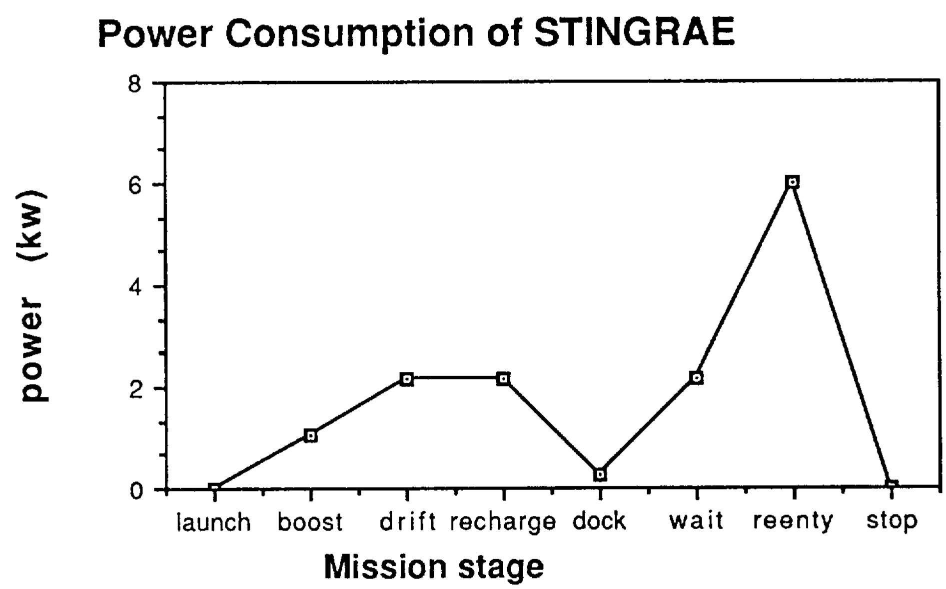


figure 20

While docked to the space station, STINGRAE will require a recharge of its main battery system and additional power for "stand-by" operation of all its subsystems. Once recharge is completed, the power drain upon Freedom will be only "stand-by" and therefore minimal, (see figure 20). The power supplied by Freedom will enter the circuit via a power cable (see figure 21a). The cable will attach to an adapter specially developed for STINGRAE which will be installed and tested prior to launch of the initial mission. Two adapters per docking area will be installed for the purpose of redundancy. After docking of STINGRAE is completed, the cable, which will be stored near the docking hatch on a motorized rapidly retracting wheel assembly, will be manually connected to the power adapter. A second cable will be stored near the wheel to be used as a replacement. The cable will be segmented (figure 21b) to allow for safe separation during rapid retraction in

the event that disconnection from the adapter is not possible, i.e. an emergency evacuation of Freedom.

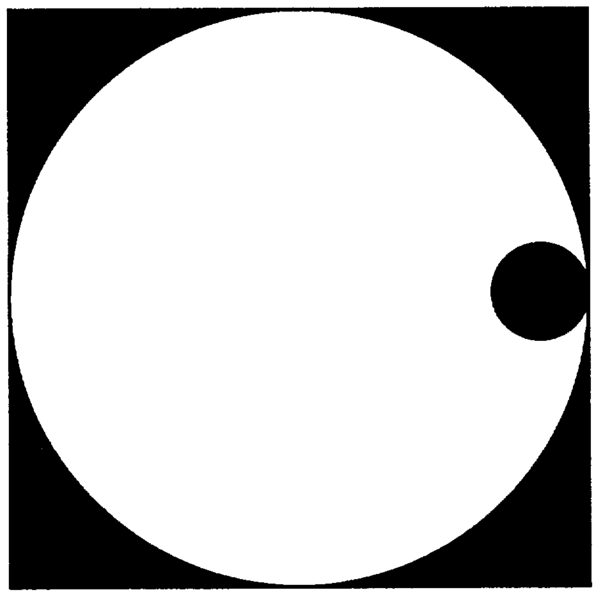


figure 21a

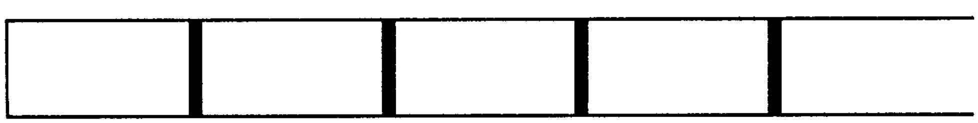


figure 21 b

The return voyage for STINGRAE begins with a check of the primary and secondary battery systems. After station separation, STINGRAE will again be operating under primary battery power. Once final orbit insertion is obtained, STINGRAE will wait for its

reentry window. During this time, all power will be supplied by the main batteries (see figure 20).

After being cleared for reentry, the secondary battery system will become activated and supply the power for reentry. This battery system is also composed of Silver - Zinc cells. During reentry, the maximum power load of the mission will occur (figure 20). The majority of power consumed during this phase of the mission will be used to steer and stabilize STINGRAE. All active control surfaces will be used during reentry.

During vehicle turn-around tests, the cable will be used to supply vehicle power. Upon delivery to launch site facilities, both the primary and secondary batteries will be recharged.

The schematic of the electrical circuit used for STINGRAE, (figure 22), displays the redundancy introduced to eliminate single failure destruction. The battery sources, both primary and secondary, have been divided in half. The two halves, connected in parallel, each posses enough storage power to complete their task under "near normal" operations. STINGRAE'S power system, as mentioned previously, is large enough to handle the longest mission time required and therefore, in the event of a single failure, would still be capable of completing the mission. The schematic also displays the redundancy of the d.c. converter and recharge regulator.

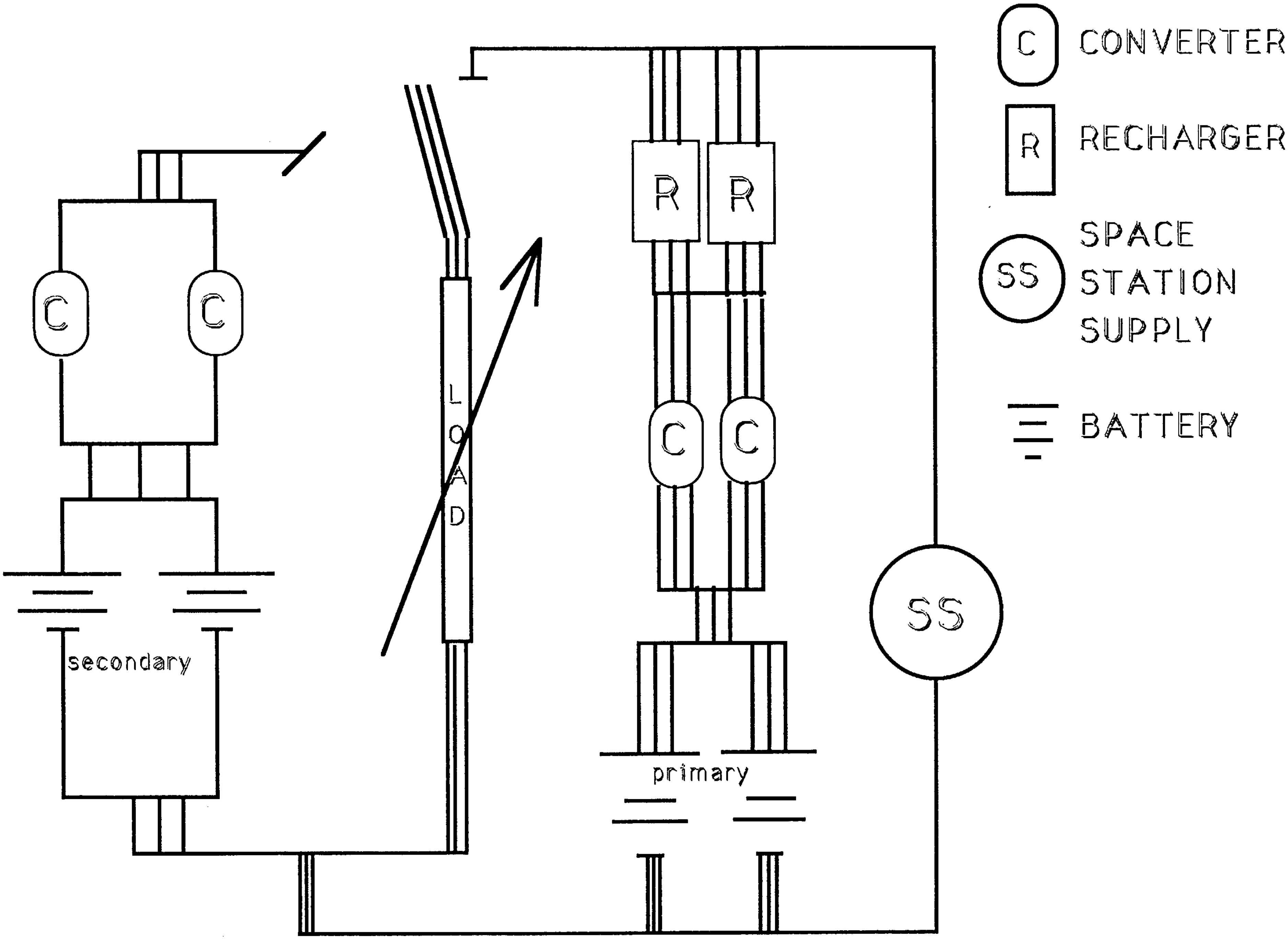

C
R
SS
CONVERTER
RECHARGER
SPACE
STATION
SUPPLY
BATTERY
L
O
A
D
secondary
primary

figure (22)

The use of Silver - Zinc storage batteries on STINGRAE was based upon a need for a large storage capability (high energy volume), low weight (high energy density), and the absence of a need for multiple discharge and recharge of the batteries (low cycle operation).

The sizing of the batteries for STINGRAE appears on the following page. The calculations for approximate volume and mass are shown. The actual dimensions of the batteries are not shown but appear under the section entitled component layout.

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Propulsion

The propulsion system of STINGRAE is required, by the RFP, to meet certain specific and derived requirements which are: to determine and produce the delta V needed to reach space station Freedom and the orbital platforms, both near and polar, to produce enough delta V for reentry, to insure against single point failures, to rendezvous with the space station under N.A.S.A. approved means (no corrosive exhaust in a "dead" zone around station), to be low cost and highly reliable, and to use off the shelf technology whenever possible.

Battery	Ag - Zn
Energy density (E. D.)	120 Wh/kg
Storage volume (S. V.)	200 Wh/L
Depth of discharge (dod)	80

Mission Requirements	
Load (Pl)	2 kW
Time (t)	16 h
*load	6 kW
*time	0.5 h

* peak values

Stored Energy (S. E.) = Pl * t / dod

40 kW-h

Battery Weight = S. E. / E. D.

333.3333 kg

Battery Volume = S. E. / S. V.

0.2 m^3

* Stored Energy

3.75 kW-h

Battery weight

31.25 kg

Battery volume

0.01875 m^3

In response to these constraints, the propulsion system for STINGRAE consists of two propulsion subsystems: a chemical system and a gas expulsion system. The propulsion system uses a modified space shuttle orbital maneuvering engine in conjunction with a forced Helium feeding system. The engine mixes nitrogen tetroxide and monomethylhydrazine to achieve a Isp, at altitude, of 325. The fuel calculations, including mass and volume per tank, as well as the necessary delta V requirements for the mission, were determined using the rocket equation and appear on the previous page.

The amount of delta V needed for reentry will be preset and will not vary from mission to mission (this calculation should appear under reentry). As a result the amount of fuel allotted for reentry will also be constant. However, the amount of propellant needed to obtain initial space station orbit is largely related to the altitude of the space station at time of rendezvous. Since this will be a variable, mission objectives will depend upon how much fuel mass is needed to obtain rendezvous orbit (see figure 25). The calculation on the following page represent attainment of a space station orbit of approximately two-thirds it maximum altitude. All further calculation, i.e. tank sizing, system mass figures, etc., will be based on this figure.

Mission Data

vehicle mass	3,200.00	kg
down mass	13,094.00	kg
up mass	16,220.00	kg
Spec Impulse	320.00	sec
DELTA V up	107.50	m/sec
DELTA V dn	315.00	m/sec

Rocket eqtn delta v = Isp * g * ln (Mi / Mf)

		1.05 times
Boost Fuel	737.29	774.15
Reentry Fuel	1,721.70	1,807.78

mixture ratio = 1.65

	N2 O4	CH3NHNH2	Helium	
Spec Grvty	1.40	0.87		
mass	1,607.62	974.31	13.65	kg
volume	1.15	1.12	1.19	m^3

Propellent
volume per tank (m^3) 0.29

TOTAL

2,581.93 Kg
2.27 m^3

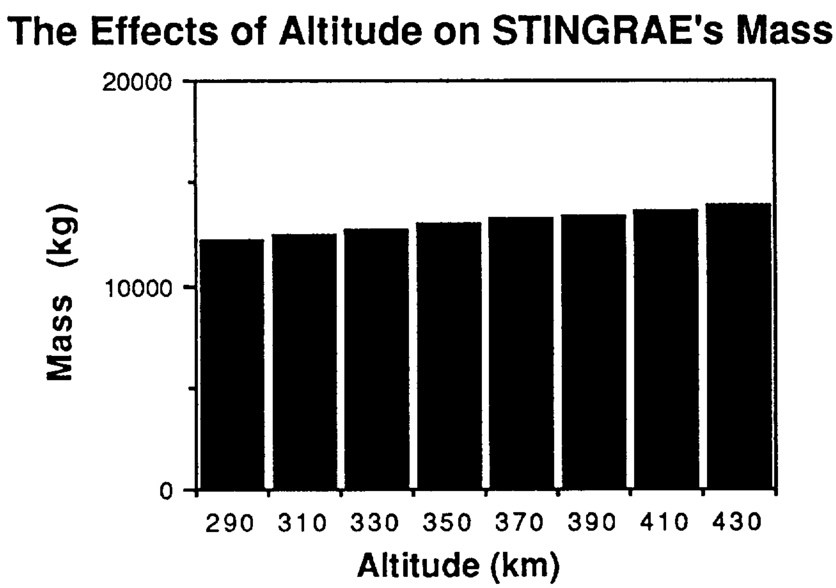


figure 25

Since the exact altitude of Freedom will be known ahead of time, mission schedules and specification can be properly altered. In the event less fuel is needed for a given mission, the tanks will simply be partially filled. In the event more fuel is needed, additional tank, half sized, will be employed.

The calculation for the sizing of the propellant tank and the feeding system appear on the following page. The tanks are cylindrical in shape and have a diameter and height as shown. The external volume and number of each tank appear boxed at the bottom of the page. The helium tanks are made out of aluminium while the propellant tanks are constructed of an internal tank of titanium and an external tank of aluminium. The propellant tanks were design to save weight over an all titanium tank and to protect against

Engine Data	
mass (kg)	100.00
thrust (N)	26,689.00
exit v (m/:	8,000.00

Tank Data	
P1 fuel tank	1,100.00 psia
P2 fnl gas =	1,150.00 psia
P3 intl gas :	4,000.00 psia
Temp =	520.00 F
He gas cons	386.00 ft/ F
Vol fuel	20.34 ft^3
gamma He	1.34

Feeding Tanks	
radius (R)	0.50 m
height	1.50 m
diameter	1.00 m

Propellant Tanks	
radius	0.25 m
height	1.50 m
diameter	0.49 m

Thickness eqtn. = Pt*R / (YS / SF - Pt / 2)

tank pres	27579029 Pa	tank pressure (Pt	7584233 Pa
Yield strgth	5.5E+08 Pa	Yield strgth	1.7E+08 Pa
safety factoı	2.00	safety factoı	2.00
tank thickns	0.05 m	tank thcknss	0.02 m
	1.55 m^3	tank vol	0.35 m^3
	1026.76 kg	tank mass	200.98 kg
4 tanks @ kg	2,053.52	8 tanks @ kg	1,607.86

corrosion of an all aluminum tank. While the actual layout of the system, tanks and engine, appears in the section on component layout, a diagram of the entire system appears under the title of **Propulsion System.** The system employs single point failure protections and uses multiple storage tanks to insure against contamination. A fuel mass of 1.05 percent is also used to insure enough fuel is present. The pipes connecting the tanks are assume to display Hagen-Poiseulle flow and this is accounted for through pressurizing propellant tanks to 1100 psi instead of 1000 psi.

No delta V calculation are shown for platform maneuvers since STINGRAE will not be going to the platforms. The following page contains calculation as to how much fuel would be required for STINGRAE to complete a mission to the polar platform. The velocity changes necessary and their accompanying mass requirements make this requirement infeasible. N.A.S.A. already has plans for an orbital transfer vehicle, OTV, to assist the space station. It is therefore STINGRAE policy that all platforms be brought to Freedom by the OTV's and resuppied by space station personnel independent of STINGRAE, i.e. space walk or mechanical arm.

As mention above, the engine of STINGRAE is a scaled down version of a shuttle's orbital maneuvering system. The scale down is in reference to the amount of times the engine is designed to fire (500 min.). STINGRAE's engine will fire one order of magnitude less as many times. The scale down of this aspect of the shuttle engine

ORBIT CALCULATIONS

u	398600.00
angle difference	1.21
a	6,668.14
	6,688.14
altitude	6,708.14

V of space station orbit 7.73 km/sec

orbit change to polar platform from space station

Equation delta V = 2*V sin (O/2)

= 8.81 km/sec needed to obtain same plane as polar platform

Now an altitude change is needed

delta V = Vneeded - Vhave

Vneed = (u*(2/r1-1/a))^.5

= 7.74 km/sec

delta V = 0.01 km/sec

Vneed = (u*(2/r2-1/a))^.5

= 7.70 km/sec

V2 need = (u/a)^.5

= 7.71 km/sec

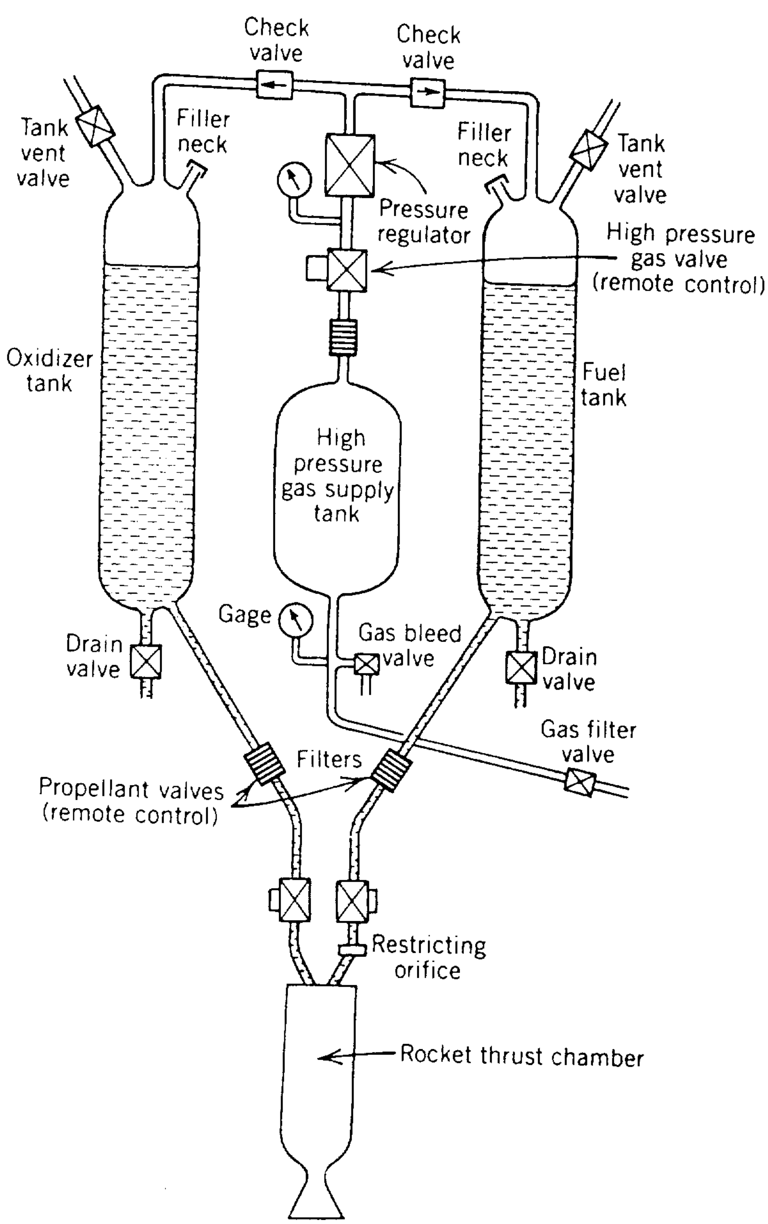
delta V2 = 0.01 km/sec

Total delta V 0.02 km/sec

Therefore the total delta V needed for entire trip (to and from) is double the sum of the total delta V's = 17.67 km/sec

Using the rocket equation an Isp of 400 (O & H) and a LRM mass of 6000kg mass = 248792.00 kg

PROPULSION SYSTEM



is hoped to drive down STINGRAE's engine cost. In all other aspects, the two engine should be the same.

The feeding system will use pressurized gas, helium, to displace the propellants. This type of system has been extensively used in space and is a simple and reliable means of throttling an engine. A gas feed system also eliminates chugging of fuel. A feeding system is paired with each propellant tank and several feeding lines and valves are incorporated to insure redundancy.

The fuel will be mixed at a 1.65 ratio (same as shuttle's engine). It has been widely used in space and can be stored for long durations in such tanks as described above. Nitrogen tetroxide and monomethylhydrazine possess a high Isp and are hypergolic, therefore requiring no starting mechanism.

The second propulsion subsystem uses force cold nitrogen. This subsystem is used to maneuver STINGRAE in the "dead" zone around Freedom. The system also doubles as a attitude and articulation system and further details of the system can be found under the same heading.

References

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Group #3
LSCS- Deidre Caldwell

<u>Life Support and Crew Systems</u>

The purpose of the life support and crew systems (LSCS) is to provide the necessary essentials for a crew's survival and comfort in a manned spacecraft vehicle. Designing for the crew's requirements is relatively complex in terms of the biological and engineering aspects that have to be taken into account in order to maintain an efficient as well as comfortable life support system. The design requirements can be broken into three main divisions of control and management: (1) environmental, (2) water and (3) waste.

The environmental control entails many requirements. A shirt-sleeve environment is needed for the crew members for comfort. With the design of an efficient LSCS, there should be no need for a continuous use of a space suit. However, space suits will be provided in case of an emergency. The need for supplies of the atmosphere such as nitrogen and oxygen must be in abundance for at least 24 hours use in space in conjunction with the other consumables (lithium hydroxide, food, and water). There should also include a cooling system for the metabolic and avionics heat loads that are generated within an enclosed system. Fire detection and suppression are important for human safety considerations. The lithium hydroxide system will provide removal of carbon dioxide and contaminants from the cabin's atmosphere.

Another system, water control and management, entails providing water for drinking and sanitation purposes by storing the water in the cryogenic tanks. In addition, the disposing of the waste water (from water vapor) has to be taken care of in the space vehicle.

Thirdly, waste control and management disposes of all the wastes that has accumulated on the vehicle. The wastes includes human solid and liquid wastes, uneaten food and expendable solid wastes such as wet wipes, plastic gloves and liner bags. These wastes are placed in a container and later removed after the mission has completed.

Along with the above mentioned requirements, other factors have to be considered to perform the project objective of STINGRAE.

These factors are: (1) storage of foods, (2) medical supply and (3) living space provisions. In regard to LSCS, one concern is to safely return the crew members back to earth from the space station in an emergency event. Thus, the following factors have to be taken into account: (1) reasons for leaving the space station, (2) fail safe redundancy and (3) equilibrium with the space station environment.

To design the life support and crew systems, one vital aspect is the duration and the number of passengers participating in the mission. In order to determine an appropriate length and number of men, trade studies and engineering analysis were made with mission planning. The results for project STINGRAE are:

1. Number of Crew/Passengers	6 men
2. Mission Length/Duration	24 hours (1 day).

This duration is not the time of the return to earth from the space station. It is the time allotted for providing consumables for the crew members in case of trouble occurring when returning to the planet earth.

With such a short mission duration, it would not be practical to consider a regenerative system for LSCS. The crew will be aboard the vehicle only in emergency situations; otherwise, the vehicle will be used as a logistics resupply transporter for the space station. Taking this into consideration, there will be no reasons for intricate designs for a kitchen galley, sleeping stations or urinal water-flushing systems like that of the Space Shuttle Orbiter.

System Description

Environmental Control System

The single-gas system such as oxygen would be more easier to control than a dual-gas system. However, the major disadvantage of a single-gas system is that pure oxygen is a fire hazard. Thus, STINGRAE is pressurized with 21% oxygen and 79% nitrogen designed to operate at 101.325 kN/m^2. The cabin pressure is maintained by means of a regulator. In case of an emergency, the regulator can be

turned off and another regulator will support the cabin at 55 kN/m^2 similarly to the Space Shuttle Orbiter.

The pressurization system consist of one oxygen tank and nitrogen tank system. For each of the consumables (oxygen, nitrogen, lithium hydroxide and water), cryogenic tanks are used for storage because they are condensed, light weight, and thin-walled. Figure A.1 is a listing of how the volume sizing of the tanks where calculated. A trade study (Figure A.2a) was done with three metals that would be acceptable to store the consumables: steel, aluminium and titanium. Figure A.2.b is a table of the density, yield strength and mass values of the three materials. The aerospace material used is aluminium 2024-T4. This metal has low density of .1000 lb/in^3 which constitutes a lightweight mass for the storage tanks. Aluminium was an appropriate choice due to the considerable weight savings which in turn reflects a cost reduction compared to the other materials.

The oxygen tanks are pressurized at 20678.6 kN/m^2 by controlled heaters and released into the cabin area in a gaseous form to the oxygen supply valve. This gaseous oxygen flows through a cabin heat exchanger where the gas is warmed before passing through the regulators. Two tanks, where one is used for emergencies (50% reserve), are provided.

The nitrogen system has two storage tanks (one for reserve) at 20678.6 kN/m^2 (Figure A.3b). Similar to the Space Shuttle Orbiter, the nitrogen valve controls the pressure of the nitrogen gas to 1378.6 kN/m^2 when it arrives at the regulator. Then, the nitrogen is joined with the oxygen by a control valve. The nitrogen/oxygen pressurization system will provide airflow into the cabin by means of vents and inlets. If the inside air pressure is lower than the outside pressure by 1.4 kN/m^2, the vent valves will be opened to permit air to flow into the cabin. In addition, these valves can be made to emit air from the cabin when the cabin pressure exceeds 107 kN/m^2.

The air circulation is provided by a cabin fan (an additional one is used for emergencies). It operates much like the Space Shuttle Orbiter by propelling air from the cabin to the lithium

Figure A.1

This is a listing of the equations used in calculating the total mass, height and diameter of each type of cryogenic tank.

Unit conversion 1	$1\ ft^2 = 144\ in^2$
Unit conversion 2	$.02832\ m^3 = 1\ ft^3$
Unit conversion 3	$6892.857\ N\text{-}in^2 = 1\ lbf\text{-}m^2$
Unit conversion 4	$0.4535\ kg = 1\ lb$

gas constant : R [ft-lbf/lbm-°R]

oxygen: R= 48.28

nitrogen: R= 55.15

m= molecular weight
Ro= universal gas constant
R= specific gas constant
= Ro/m

water: R= 85.772

m= 18.016 g/mol
Ro= 8.3144 Joules/°K-mole
R= (8.3144 J/°K-mol)(mole/18.016g)(.737652ft-lbf/J)
(1 g/.0022046 lbm)(1 K/1.8 °R)

LiOH: R= 64.52

m= 23.95 g/mol

temperature : T= 540 °R
pi constant : π= 3.14159
density : p [lbs/m^3]
mass : m [kg]
tank pressure : P [N/m^2]
inner volume : Vi [m^3]

$Vi = mRT/P$

m [lbm], R [ft-lbf/lbm-°R], T [°R], P [lbf/in^2] and using unit conversion 1 & 2.

inner radius : ri [m]

$ri = \sqrt{(Vi/\pi hi)}$

hi [m] = assign an arbitrary value; by changing the value of hi, the mass of the tank can be adjusted to reach a desired mass.

yield strength : Sy [N/m^2]
Factor safety : Fs

$Fs = 2$

stress : s [N/m^2]

$s = Sy/Fs$

tank thickness : t [m]

$t = Pri/(s-P/2)$

outer height : h [m]

$h = hi + 2t$

outer radius : ro [m]

$ro = t + ri$

outer diameter : do [m]

$do = 2(ro)$

outer volume : V [m^3]

$V = \pi(ro)^2h - \pi(ri)^3$

tank mass : m1 [m]

$m1 = pV$

mass total : mt [m]

$mt = m + m1$

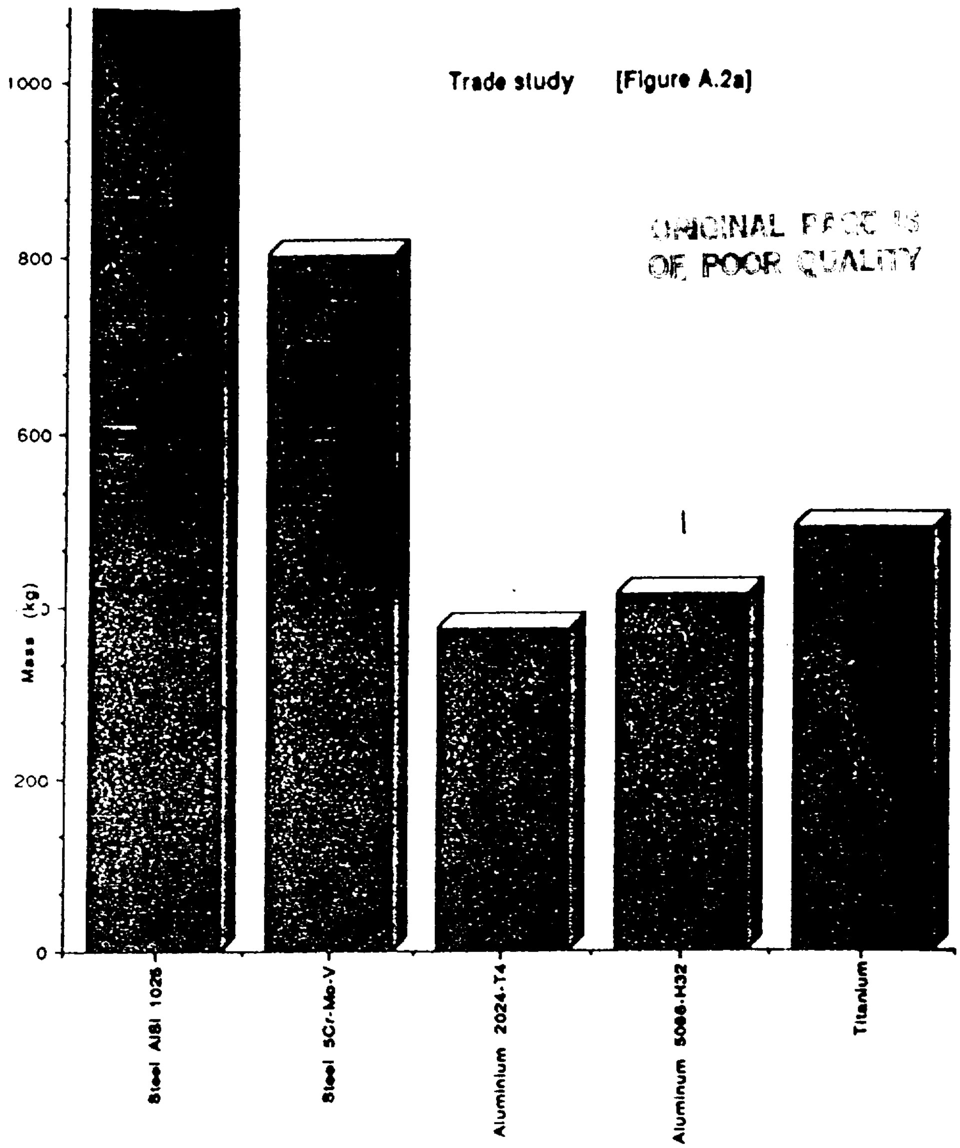


Figure A.2b

Material Type

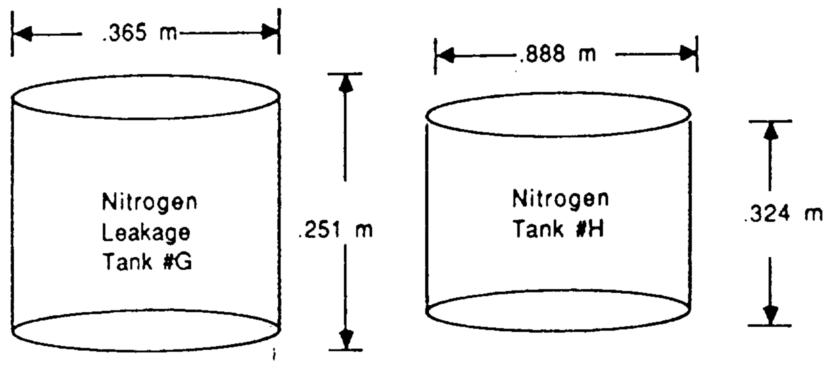
This is a table of the density, yield strength, and mass values of steel, aluminum, and titanium. These factors were determined by the same process displayed in Figure A.1. From this data, a trade study was compiled to determine the most desirable material type in terms of weight considerations. The substance stored was water at a pressure of 20678571 N/m^2 and temperature of 540 °R.

Additional constraints:
m= 6.6025 kg
h= 2 m
R = 85.772 (water) ft-lbf/lbm-°R

Material type	density [lb/m^3]	yield strength [ksi]	mass [kg]
Steel			
AISI 1025	17328.81	36	1092.595
5Cr-Mo-V	17145.76	200	804.9046
Aluminum			
2024-T4	6101.69	40	375.493
5086-H32	5857.63	28	414.0367
Titanium	9884.75	110	492.14

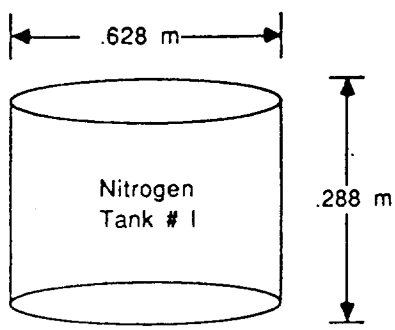
Figure A.3b

This is the mass and dimensional factors for the nitrogen tanks. Sample calculations are provided.



mass (nitrogen)=3.59 kg
mass (tank + nitrogen)=
89.43 kg

mass (nitrogen)= 21.29 kg
mass (tank + nitrogen)=
176.45 kg



mass (nitrogen)= 10.64 kg
mass (tank + nitrogen)=
176.11 kg

Material type: Aluminum (cryogenic)
Consumable type: Nitrogen (leakage)
Tank ID: #G

Unit conversion 1	144	in^2/ft^2
Unit conversion 2	0.02832	m^3/ft^3
Unit conversion 3	6892.857	N-in^2/lbf-m^2
Unit conversion 4	0.4535	kg/lb
gas constant	55.15	ft-lbf/lbm-°R
temperature	540	°R
pi constant	3.14159	
density	0.1	lbs/in^3
density	6101.695	lbs/m^3
mass	7.92	lbm
mass	3.59172	kg
tank pressure	3000	psi
tank pressure	20678571	N/m^2
inner volume	0.015462	m
inner height	0.2	m
inner radius	0.156873	m
Yield strength	40	ksi
Yield strength	275714.3	N/m^2
Safety Factor	2	
stress	1.38E+08	N/m^2
tank thickness	0.025439	m
outer radius	0.182311	m
• outer height	0.250878	m
• outer diameter	0.364623	m
outer volume	0.014068	m^3
• tank mass	85.83991	kg
• total mass	89.43163	kg

hydroxide canisters. These canisters have to replaced on a daily basis. The main function of the canisters is to remove non-metallic materials, stored gas leakage, metabolic processes from the crew, odors and contaminants. The canisters contain a layer of activated charcoal, glass wool filter, and lithium hydroxide. The activated charcoal absorbs the odors and noxious gases. It absorbs organic materials such as alcohols and hydrocarbons. A glass wool filtration minimizes the aerosol hazards such as Freon 1301(fire extinguisher chemical). It will also trap the solid particles and lithium hydroxide from entering the cabin's atmosphere. The carbon dioxide is removed by means of the lithium hydroxide. This substance is highly reliable and readily absorbs carbon dioxide in the presence of water vapor in the gas stream. The exothermic chemical reaction in Figure A.4 illustrates this principle. If the carbon dioxide is not removed, the crew will suffocate. Thus, the present design levels for the carbon dioxide partial pressure is 0-8.0 mmHg for normal design limits and 0.3 mmHg as an optimum value.

The cabin temperature is maintained at 70° - 75° F by use of manual temperature controllers. To regulate the humidity, the air flow pulled over the coldplates (heat sinks or special metal plates that contains channels through which water and mixtures flow) from the cabin heat exchanger. Condensation occurs when the temperature changes as the air flow passed over the coldplates. A centrifugal water separator, fans and the cabin heat exchanger divides the water from the air. The air is recirculated back into the cabin; whereas, the water is vented overboard. An air circulation system for the orbiter removes 1.8 kg/hr of water.

Besides circulating the desired temperature and air mixture, the air circulation system also collects the heat from the crew and crew avionics. Warmed cabin air is passed through the cabin heat exchanger and the excess heat is directed to the water coolant loop. For STINGRAE, the amount of heat released from various system can be viewed in Figure A.5.

The water coolant loops have pumps that pass the water and heat through a Freon interchanger and then to the radiator. Because of the high latent heat of vaporization and the absence of pressure,

water can boil at low temperatures. The radiator and the flash evaporators will boil the water at low temperatures and pressures. Then, the outcoming steam vapor is vented out to space by means of a cabin pressure relief valve.

Fire is detected by means of smoke detectors, which are distributed throughout the space vehicle. The smoke detectors will alarm when any type of increase of gas or combustion occurs. The fire extinguishers will be used for suppression of the fire. Bromotrifluromethane or Freon 1301 is used for chemical fires because instead of smothering the fire, it breaks down the chemical reaction of the fire. Figure A.8 has a listing of the number of extinguishers used in LSCS.

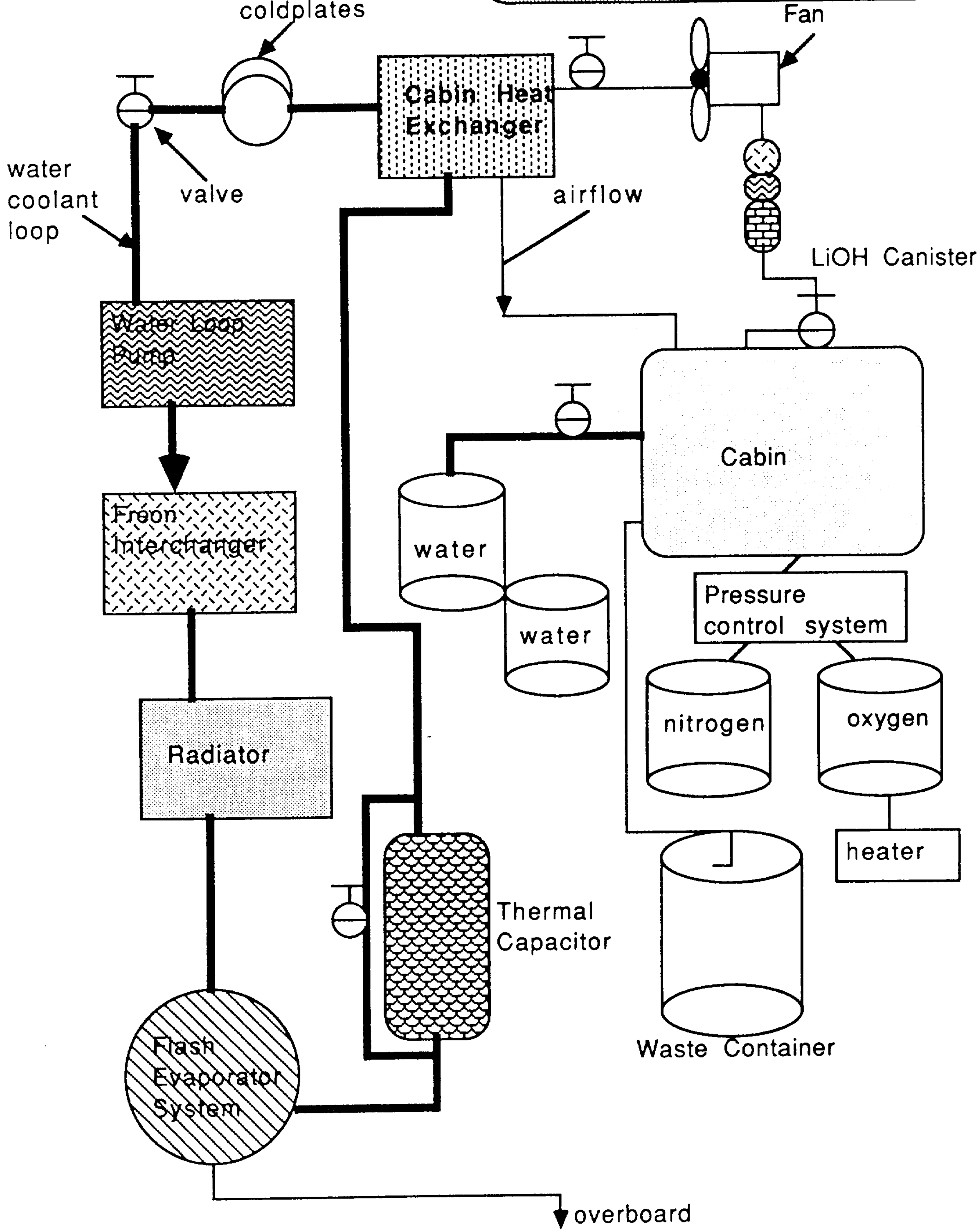
Figure A.7 is a schematic diagram of the environmental LSCS system loop. Figure A.8 is a listing of the component's dimensions and power values.

Water Control and Management System

The water system is one of the most critical life support requirements. Because of the duration of STINGRAE, a pressure control regulator will monitor the water flow from the cryogenic water tanks to a water control valve. Even though the Space Shuttle Orbiter provided water from the by-products of the fuel cells, it would not be advantageous for STINGRAE in terms of extra weight of pumps and valve. A microbial check valve and filtration system is located in the supply line between the cryogenic tank and the water dispenser. The dispenser will be used to allow the crew member to gather the amount needed for drinking. Once the water tank is empty, a water meter will signal the attachment of another water tank. This will be done manually by a crew member. The water's temperature will be the same as when it was stored inside the cryogenic tank. There will no devices for adjusting the temperature of the water.

The collection of waste water that has been drawn from the atmosphere is vented overboard in the form of steam by use of the radiator and flash evaporator.

Figure A.7



coldplates
Fan
Cabin Heat Exchanger
water coolant loop
valve
airflow
LiOH Canister
Water Loop Pump
Cabin
Freon Interchanger
water
water
Pressure control system
nitrogen
oxygen
Radiator
heater
Thermal Capacitor
Flash Evaporator System
Waste Container
overboard

Figure A.8

Listed below is the mass and dimensional factors for the components in LSCS. Because of insufficient data, all the measurements could not be located. Most of the data was gathered from the ECLSS of the space shuttle. The shuttle is quite larger than our vehicle. However, most of the area is used for storage and the mid deck area is very spacious. On the other hand, our vehicle's goal is to decrease the mass and volume specifications. Taking this into consideration and the fact that the number of men on the orbiter is similar to our vehicle, the dimensions and power constraints of the orbiter was reduced by a factor of 1/2 in order to get the measurements for the STINGRAE vehicle.

System	Number	Mass (kg)	Height (m)	Width (diam.) (m)	Length (m)
Tank A	1	599.2	.435	.899	
Tank B	1	390.9	.398	.635	
Tank C	1	146.5	.358	.344	
Tank D	1	78.38	.344	.243	
Tank E	1	166.7	.283	.595	
Tank F	1	109.7	.259	.420	
Tank G	1	89.43	.251	.365	
Tank H	1	176.45	.324	.888	
Tank I	1	176.11	.288	.628	
Freon 1301	4	6.35			
Fire Exting.	4	34.36	.8128	.2286	

Note: The height, length and width are the same.

System	Mass (kg)	Length (m)	Power (Watts)
Cabin Heat Exchanger	9.96	1.35	
Coldplate Waterloop	46.67	.1074	
Cabin Temp. Controller	2.22	.1723	8
Heaters(2)	.1134	1.20	6.67
Flow sensors(2)	.374	.44	
Pressure sensors(12)	1.02	.0801	0.5
Carbon dioxide sensors	1.21	.0108	0.1
Water Bypass controller	2.23	1.148	4.0
Main Cabin Fan	2.04	.1367	90
Fan Downstream Valve	.102	.0775	
Venting Fan			8.5
Bypass valve	1.15	.261	3
Waterloop Pump	7.24	.261	98.5
Water bypass valve(3)	1.93	.142	4.35
Flash evaporator system	13.13	.352	4
Thermal Capacitor	45.36	.384	
Food and Containers*	5.44	.291	
FES Duct Heaters			12.5
Fire Suppresion			11.5
O2/N2 Supply Panel			2.25
O2/N2 Control Panel			225.12

** Note: Food Calculation*
food consumption: 1.5 lb/man-day(6 men)(1 day) = 4.08 kg
expendable containers: 0.5 lb/man-day(6 men)(1 day) = 1.36 kg
Total = 5.44 kg
density of food as packed for storage = .008 lb/in^3
volume= .0245 m^3; *length*= .291 m

Waste Management and Control System

This system collects human wastes in addition to wastes from food and other paper-like material. Within the area designated as B-Room (Figure A.6), a crew member can release his wastes (feces and urine) into a plastic, durable, water-proof bag located in the center of the commode assembly. Restraints for the feet and waist and a handholds are situated for the passengers positioning and stabilization when using the B-Room. The toilet tissue, waste and germicide are sealed in the plastic pouch and then stored in a trash container. The germicide kills the microorganisms that causes the decay and odor. In addition, a vent will be located in the B-Room for the removal of odors and gases. The tissue is a multi-ply, absorbent and low-linting paper material. The crew member should then clean the seat of the commode with a biocidal cleanser and a general purpose wet wipe while disposable plastic gloves are worn. These items are placed in a plastic bag and stored in the trash container. A newly bag liner should then be placed in commode seat assembly. Wet wipes (personal hygiene miniature towels that contain quaternary compound ammonium), uneaten food and miscellaneous trash are disposed in a plastic, water-proof bag in the trash container. A privacy curtain of Nomex cloth is attached to the walls which isolates the B-Room from the rest of the cabin area.

The trash container has a liner and must be fastened. It is located in a separate storage area and it includes a ventilation system.

Food Management System

The quality and quantity of food consumed by the crew members of the space vehicle should approximate closely to a normal diet as on earth. The food will be freeze-dehydrated and bite-sized compressed. Since water is removed from the food by this process without damaging or changing the chemistry, about 70% of the bulk weight can be reduced. The food will be consumed directly from the package. The packages are made of laminated plastic bags that are over-wrapped in a non-flammable flurohydrocarbon. No oven or refrigerators will be needed in order to

reduce weight. However, utensils, mainly plastic spoons, will be provided so that a crew member can eat right out of the plastic pouch.

Medicine Supply

Because many possible crew illnesses and injuries will occur on the space station, STINGRAE must be able to accommodate for such situation. However, X-ray machines and clinical laboratories are not feasible in terms of volumetric considerations for STINGRAE. Only the basic medical equipment should be placed on the spacecraft. Figure C.1 details a typical kit supplied to Gemini astronauts. For STINGRAE, these kits will provided for each crew member in addition to extra bandages, cold packs and splints.

Living Space Requirements

Establishing an appropriate volumetric standard is vital in order to consider the amount of living space available for the crew. A minimum (lower limit) of 1.42 m^3/person is adequate for 1 or 2 days of confinement where no impairment or marked impairment occurred during this brief confinement. The other limits can be calculated by the following tolerance volume requirements equations:

$$V(min) = -(0.0040)x^2 + (1.4219)x + 81.307$$
$$V(acc) = -(0.0068)x^2 + (2.8346)x + 83.440$$

where x is the known mission duration measured in days and the resultant volume is measured in ft^3/man-day. To convert the resultant volume to cubic meters, multiply by the number of men, the number of days, and .02832.

For project STINGRAE, the calculations are:

Lower limit:	V	= 8.52 m^3,
Upper limit:	V(min)	= 14.05 m^3, and
	V(acc)	= 14.66 m^3.

Figure A.6

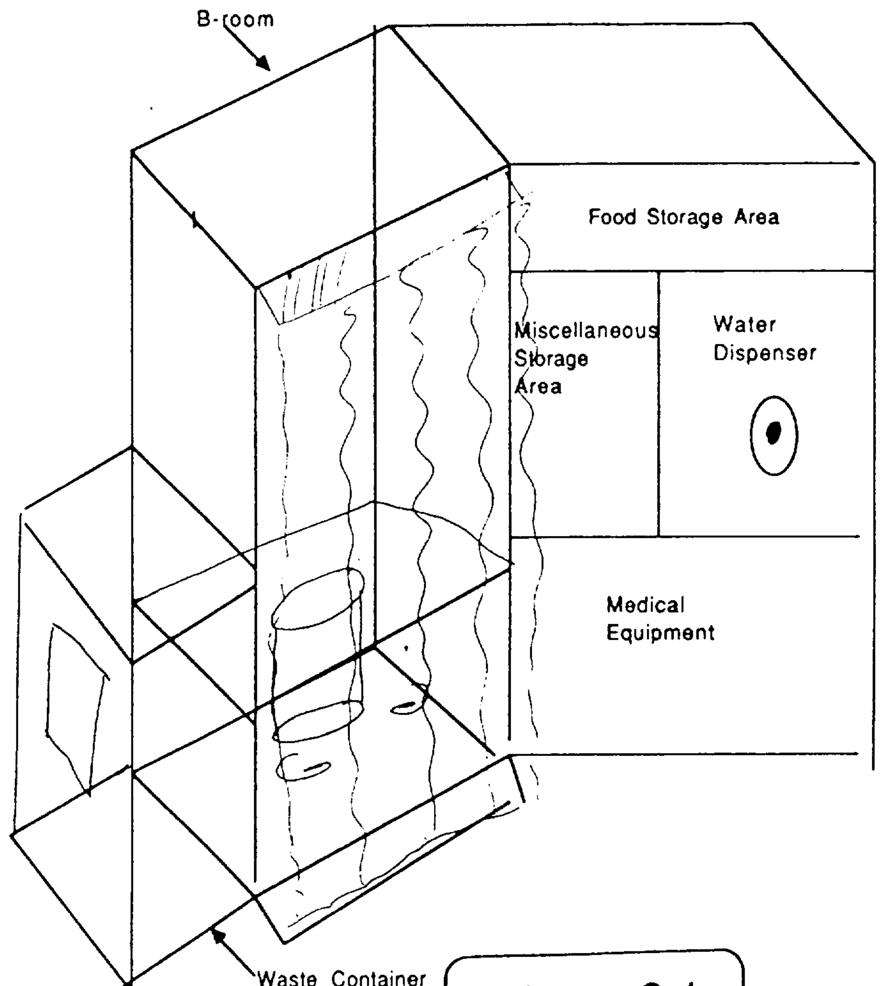


Figure C.1

Reference: Sharpe, Mitchell, R., Living in Space, Doubleday Science Series

This figure illustrates a typical emergency medical kit supplied to crew members.

Drug	Dose and Form	Use	Amount
Cyclizine hydrochloride	50 mg. tablets	motion sickness	8
Dextro-amphetamine sulfate	5 mg. tablets	stimulant	8
APC (aspirin, phenacetin, & caffeine)	tablets	pain	16
Meperidine hydrochloride	100 mg. tablets	pain	4
Triprolidine hydrochloride Pseudoephedrine hydrochloride	2·5 mg. tablets 60 mg. tablets	decongestant	16
Diphenoxylate hydrochloride Atropine sulfate	2·5 mg. tablets 0·25 mg. tablets	diarrhea	16
Tetracycline hydrochloride	250 mg. tablets	antibiotic	16
Methylcellulose solution	15 cc. in bottle	eyedrops	1
Parenteral cyclizine hydrochloride	45 mg. (0·9 cc. in injector)	motion sickness	2
Parenteral meperidine hydrochloride	90 mg. (0·9 cc. in injector)	pain	2

Threats

There are many reasons for crew members to evacuate the space station. For instance, if a fire were developed on the space station and could not be suppressed, the crew members would need an emergency vehicle to transport them to safety. Below is a listing of the possible threats and their causes:

1. Fire
2. Biological (toxic) contamination
   a. experiment
   b. fire
   c. fuel leak
3. Injury/Illness
4. Explosion/implosion
   a. leakage
   b. ruptures/structural failure
   c. relief valve fails to close
   d. fire/overtemperature
   e. chemical reaction
5. Loss of pressurization
   a. puncture
   b. inadvertent crew action
   c. internal/external leakage
   d. remove contamination
   e. fire control
   f. maintenance
6. Meteoroid and debris penetration
   a. tracking of 1-4 cm of meteorites and debris
7. Tumbling/loss of control
   a. pressure vessel penetration
   b. thruster stuck on or off
   c. collision
   d. CMG failure
   e. power failure
8. Out of control EVA astronaut
   a. fire
   b. illness/injury

c. impact
d. explosion
e. penetration
f. depressurization
g. consumables depletion

9. Consumables depletion
   a. leakage
   b. contamination
   c. LRV failure
   d. launch vehicle failure
10. Orbit decay
   a. thruster failure
   b. no fuel

With these possible threats in mind, the STINGRAE should be able to separate from the space station rapidly, availability of pressure suits, ease of entry to the earth's atmosphere, recycling of air, low-g reentry, close landing to medical facilities and the ability to track an EVA astronauts. The other subsystems will be able to provide these requirements for a safe and comfortable landing to earth. In addition, injuries, illness and uncontrollable EVA astronauts are the only causes for a partial evacuation. The other causes will lead to a total evacuation. The evacuation options that STINGRAE will be able to explore are: (1) return to earth or (2) orbit until the space station is habitable.

Other requirements

Fail safe redundancy is an important factor because it ensures that nothing will go wrong if there are backup systems or continous monitoring of the various components of LSCS. If everything is redundant and fail safe, then nothing should go wrong. The STINGRAE has sensors and meters to alert the crew if a potential problem occurs with a valve, pump, or a ventilation system. A basic tool kit will provided for the crew. In addition, a

manual override is provided for each subsystem in case a system is not working properly.

When the STINGRAE is docked to the space station, it must have similar systems so that the vehicle can use the space station's system. The only LSCS systems that will feed off of the space station is the environmental control and the power-generated systems.

Conclusions

The life support and crew system can be designed by utilizing many different combinations of design parameters. Foremost, the vehicle must provide safety for the crew. By selecting optimum subsystems to meet all of the requirements is no assurance that the LSCS will be an optimum system. Taking into consideration that the mission is not for a long duration, the vehicle need not to be a duplicate of a well-designed apartment. By minimizing the cost constraints, the size and the weight of the components in the LSCS have to be kept a minimum. Thus, only the necessary essentials for survival are needed and implemented in the STINGRAE LSCS design.

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ED HEINEN

MISSION MANAGEMENT, PLANNING AND COSTING

REQUIREMENTS:

- Identify Payloads
- Integrate Payloads into transport module
- Launch vehicle selection
- Trajectory options
- Mission Support

After receiving the requirements set forth in the request for proposal, they were organized according to importance. For mission planning purposes the two most important requirements were the establishment of payloads that would ride aboard the vehicle, and the selection of an expendable launch vehicle to lift both vehicle and payload. Because of their importance these two topics were dealt with first. Establishment of payloads was the first task to be attacked since a payload weight and volume were needed to obtain an idea of which launch vehicle could be used.

PAYLOAD IDENTIFICATION

Since the need of the space station for periodic resupply was the impetus behind the formation of our program, it is natural to ask what types of supplies are necessary for the station. Needs of the station were divided into the following categories: crew, station, and customer support. Crew support entails the replenishment of food, hygienic materials, medical supplies, and clothing. Station support involves provisions necessary for housekeeping, waste management,

trash, spares, ECLSS fluids, and EVA support. Finally, support for the customers must be considered due to the needs of the individual modules which are supplied by the customer. Needs of the customer fall into the categories of servicing plant, animal, and human research along with various other scientific experiments. Once the areas in which these supplies were going to be used was determined, it was then necessary to determine quantity and form of the supplies. Quantities and forms obtained by using data compiled from the NASA Annual Resupply Mass Summary and the OSSA Missions Waste Inventory Database were then tabulated to give the ninety requirements for up/down mass and up/down volume. These lists further itemized the resupply requirements in terms of pressurized, unpressurized fluids, and propellants which was an important consideration for the structures person when deciding to pressurize the vehicle. Finally these areas were broken down even further into crew-station and customer categories. The results are as follows:

MASS FOR RESUPPLY MISSIONS

CLASSIFICATION	MASS UP(kg)	MASS DOWN(kg)
Pressurized		
crew/sta.	4148.56	3497.99
customer	4954.14	4757.39
Unpressurized		
crew/sta.	513.01	513.01
customer	4152.18	4152.18
Fluids		
crew/sta.	360.61	0.00
customer	365.14	173.73

Propellants		
crew/sta.	45.36	0.00
customer	0.00	0.00
TOTAL	**16220.92**	**13094.30**

VOLUME FOR RESUPPLY MISSIONS

CLASSIFICATION	VOLUME UP (m^3)	VOLUME DOWN (m^3)
Pressurized		
crew/sta.	14.78	11.50
customer	13.92	13.75
Unpressurized		
crew/sta.	4.53	4.53
customer	32.64	32.64
Fluids		
crew/sta.	0.45	0.00
customer	0.50	0.00
Propellants		
crew/sta.	.57	0.00
customer	1.68	0.00
TOTAL	**69.06**	**62.59**

Two types of resupply are possible for these missions. The first type of servicing is planned servicing where certain supplies are brought up in a routine manner or schedule. This involve the replacement of consumables, refurbishment, replacement of degraded systems at known times and the scheduled replacement of old systems with new ones. The other form of resupply is of the contingency type where resupply is non-routine or non-scheduled. This means that

spares must be carried onboard the vehicle in order to be prepared for random failures.

LAUNCH VEHICLE SELECTION

After establishing the masses and volumes to be lifted into orbit, attention was turned toward selecting a launch vehicle. The main launch vehicle requirement was that it had to be expendable. Using the expendable launch vehicle information supplied in class (name of source),various pieces of information were selected to represent the best characteristics of each lifting the vehicle. The criteria used for the final evaluation were: orbit and lifting capability, launch site, payload fairing size, and Delta V needed to attain various orbits. The next step in the process was to estimate the mass of the resupply vehicle.

An initial craft mass estimate was needed to determine an initial system weight so that ELV's with lighter lifting capabilities could be ruled out. Using a structural efficiency of .2(a good estimate for a small rocket) an initial estimate of 4055 kilograms was obtained for the vehicle. The procedures used to obtain this value were as follows:

$M_s/M_i = .2$ where $M_i = M_s + M_p + M_f$

M_f = Mass of fuel

M_s = Mass of structure

M_p = Mass of payload

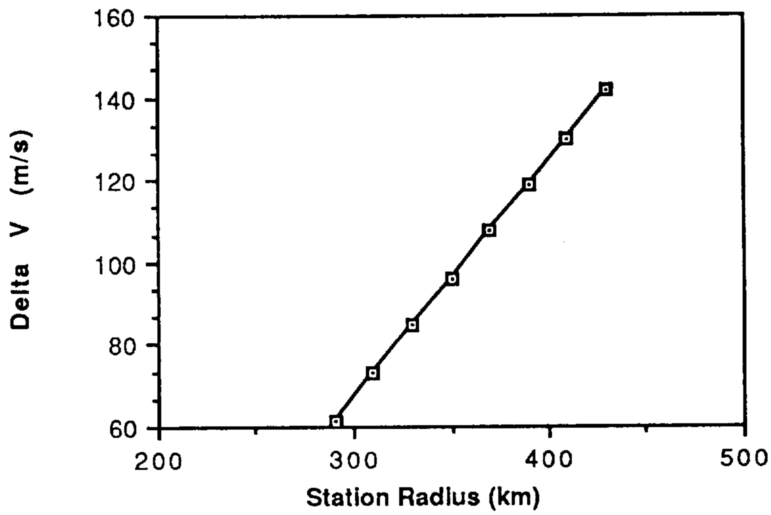
$$M_s = .2M_i = .2(M_s + M_p + \cancel{M_f})$$

$$M_s = .25M_p$$

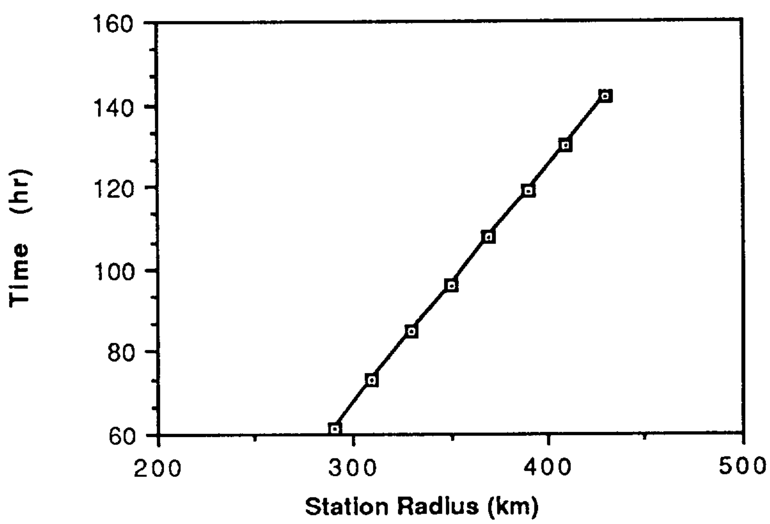
Since the fuel mass was eliminated, the resulting calculation estimates a total system mass which is lighter than the actual mass. Initially the payload mass was assumed to be equal to the full ninety day requirements in order to evaluate the possibility of a single launch fulfilling the mission requirements. However, these calculations yield system mass of 20276 kilograms without fuel. This figure cut the possible ELV's down to a Titan IV rocket using solid rocket motor upgrades(SRMU's). This version has the capability to lift 22,220 kilograms, but once fuel and tank mass were taken into consideration it was also ruled out. Therefore, the possibility of lifting the total ninety-day resupply needs in one launch was ruled out.

After learning this fact, the next step was the comparison of Delta V needed for the various space station orbits. Knowing the amount of Delta V necessary for each orbit would also helped in the development of a scheme for the allocation of mass and volume for the various launch vehicles. Data on the space station states that its orbit ranges anywhere from 290 km up to 430 km away from Earth.[1] With this information a range of Delta V's needed to achieve various station orbits were calculated based on Hohmann transfers from a 100 nautical mile orbit. The 100 nautical mile(185.20 km) orbit was used because nearly all of the possible choices for ELV's inject their cargos into this orbit. An orbit of 220 nautical miles(404.44 km) was also considered because a few ELV's which can attain this orbit. Maneuvering times from this orbit to the station orbit were also calculated for later reference in constructing the mission timeline. The resulting figures are as follows:

Delta V to station orbits from an initial orbit of 185.20 (km)



Time to obtain station orbit from 185.2 km



Below are the equations used for the calculations:

$$\sqrt{\frac{\mu}{R_2}} - \sqrt{\mu\left(\frac{2}{R_2} - \frac{1}{a}\right)} \quad \& \quad -\sqrt{\frac{\mu}{R_1}} + \sqrt{\mu\left(\frac{2}{R_1} - \frac{1}{a}\right)}$$

$$T = 2\pi\sqrt{\frac{a^3}{\mu}}$$

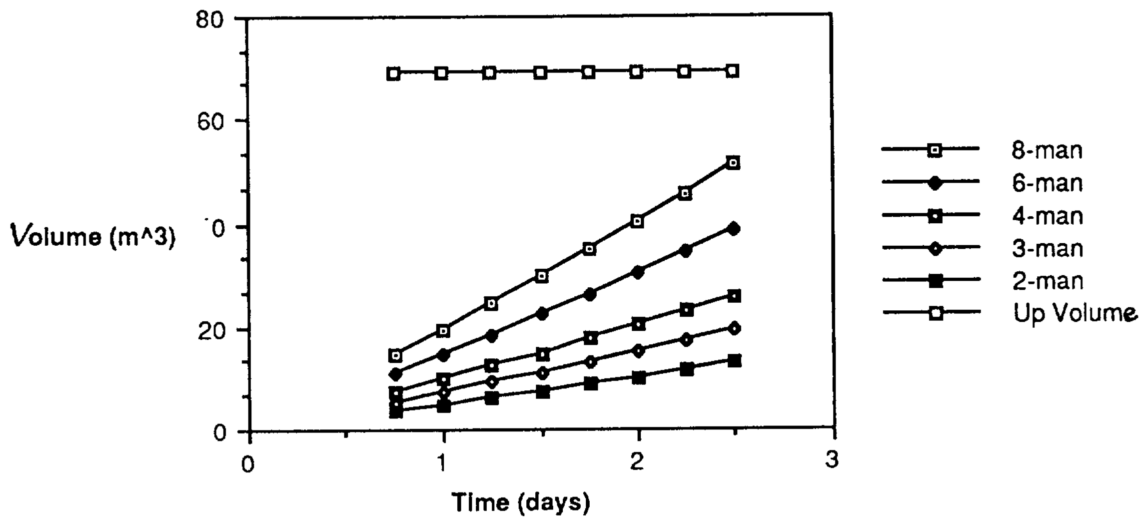
After receiving mass estimates for the various subsystems, the final percentages of the ninety-day resupply requirements to be launched each mission was determined along with the exact launch vehicle. Based on the Delta V's, cost per launch, ground support costs, and lifting capabilities the Titan IV with SRMU's which lifts roughly 22220 kg into orbit was selected. After obtaining a final vehicle weight of approximately 11000 kg, it was determined that the best percentage of the 90-day resupply requirements to be lifted each time was 50 percent.

Once the vehicle and payload sizes were determined, concentration was turned toward fulfillment of the crew emergency return requirement. The possible crew sizes were set at a minimum of two and a maximum of eight. The minimum crew size comes from the requirement that one person must always accompany an injured or ill person back to Earth. Because the space station will have at most eight people on board at a time, the crew return system need only accommodate a maximum of eight people. This poses an interesting dilemma. How many vehicles or how many people per vehicle is the optimum solution? It was immediately seen that one vehicle at the station with a capacity of eight is incapable of providing a feasible solution for an illness situation. If one person were ill, not only would another person have to accompany him but also the remainder of the crew because there would not be any vehicles to return them to Earth in case of another emergency. Likewise, a two vehicle system with each vehicle having a capacity of four people does not work.

The design of the optimum system is based on the double emergency(DE) situation of a three vehicle system where an injury or illness occurs requiring the return of a crewmember to Earth. After the vehicle has already left the station another accident occurs and crewmembers must be returned to Earth in the remaining vehicle. The worst case scenario was used in which it was assumed that one of the vehicles at the station does not work or cannot be reached. Based on this scenario the six-person vehicle is the best choice. A two-person vehicle and a three-person vehicle is ruled out because several vehicles would be necessary to cover the DE situation and thus the total cost would be enormous for launching all of the vehicles As shown earlier the four-person vehicle will also not meet the DE situation requirements.

The final choice between six- and eight-person vehicle was a lot more difficult. Both can easily sustain the DE situation. However, if two people go down in a six-person vehicle and a second emergency occurs there will be exactly six people left to ride aboard the six person vehicle; therefore maximizing the space available on that vehicle. The eight-person vehicle on the other hand would be wasting room for two extra people. If for some reason the crew cannot be returned to Earth by the normal means of transportation, a six-person STINGRAE has the maximum amount of waste carrying capability available when fully loaded as can be seen by the following graph depicting the acceptable volume for humans against time in the vehicle.

Data from "vol. dat"



Included in the graph is a line showing the volume resupply requirements in order to show that the six-person vehicle possesses the best payload capabilities at maximum crew capacities.

Overall system requirements mandate a minimum of four vehicles, one of which must be used as a test vehicle while remaining flight ready. The total number of vehicles at the station at any one times is based on the DE situation. In the event of this situation happening, two vehicles will be necessary to return the crew members and a third will be available in case access to one vehicle is denied or the vehicle is not working properly. On the ground the total number of vehicles will be four. One will be used for continuous testing and spare parts. The other three will be used in the ground-station rotation system. Once the first three vehicles are positioned at the station, the other three will be rotated in as they arrive at the station for their

scheduled delivery. Each time a new vehicle arrives, it will replace the vehicle which has been there the longest period of time. The returning vehicle will then return to Earth for refurbishment and await processing for the next mission.

PAYLOAD INTEGRATION

Once the crew size was selected, work on payload integration began. The major factor involved in arranging the payloads is whether or not they are pressurized or unpressurized. Obviously, the unpressurized items are the first items to be loaded due to the fact that they can be put into the vehicle before it is pressurized without worry of damage. Items which fall into this category are: clothing, cleaning supplies, and scientific experiments. In the same sense, some of the pressurized cargo probably will not be able to survive extended periods of time without pressurization. When live specimens are to be carried aboard the vehicle care must be taken to keep the conditions in the cargo hold at and acceptable so that they remain healthy. Medicine is another item that must be loaded shortly before launch. The astronauts cannot afford to become sick and then take medicine which is bad and worsen the situation. All unnecessary trips back to Earth are to be avoided since the major purpose of this vehicle is resupply.

Another consideration for payload integration is ease of loading and unloading supplies. Since there are a number of double racks in the space station, storage racks were developed similar to the double racks in the space station. These racks have the capability of holding the exact same drawers as used in the space station. The vehicle racks are stocked such that all drawers that must go in the same rack on the

station are also in the same storage rack on the vehicle, once again allowing the payload to be more efficiently loaded and unloaded. Still another way to increase loading and unloading efficiency is to set standard sizes on the shapes of the containers which hold the cargo. The drawers for the station double racks have already set a standard size for many objects. In order to maximize the available volume, containers which conform to the shape of cargo hold were selected. These containers are used for the storage of clothing and nonperishable items. Since many different sizes of payloads need to be carried, it is not possible to require that all items be put into standardized containers. The standardized containers will start in the rear of the craft and work forward. Some items such as the racks holding the drawers for the station will be on every mission so the loading is done using them as a starting point. This means other standardized containers will be stacked in and around the drawer racks. The further organization of the remaining items will be based on the need to balance the load around the center of mass. To do this, an inertia resolving program is used to find the new moments of inertia and center of mass for the vehicle based on the various loading schemes. The final loaded configuration is determined a couple of months ahead of time since the exact payload manifests are to be submitted several months ahead of their predetermined launch time.

Finally, human cargo must also be accounted for due to the fact that an emergency situation most likely means that they have to ride aboard the vehicle back to Earth. There are two possible positions in which the astronauts might have to return to Earth. One is in a sitting

position, and the other is in a lying position. In order to satisfy both needs the following chair in was designed: Next page

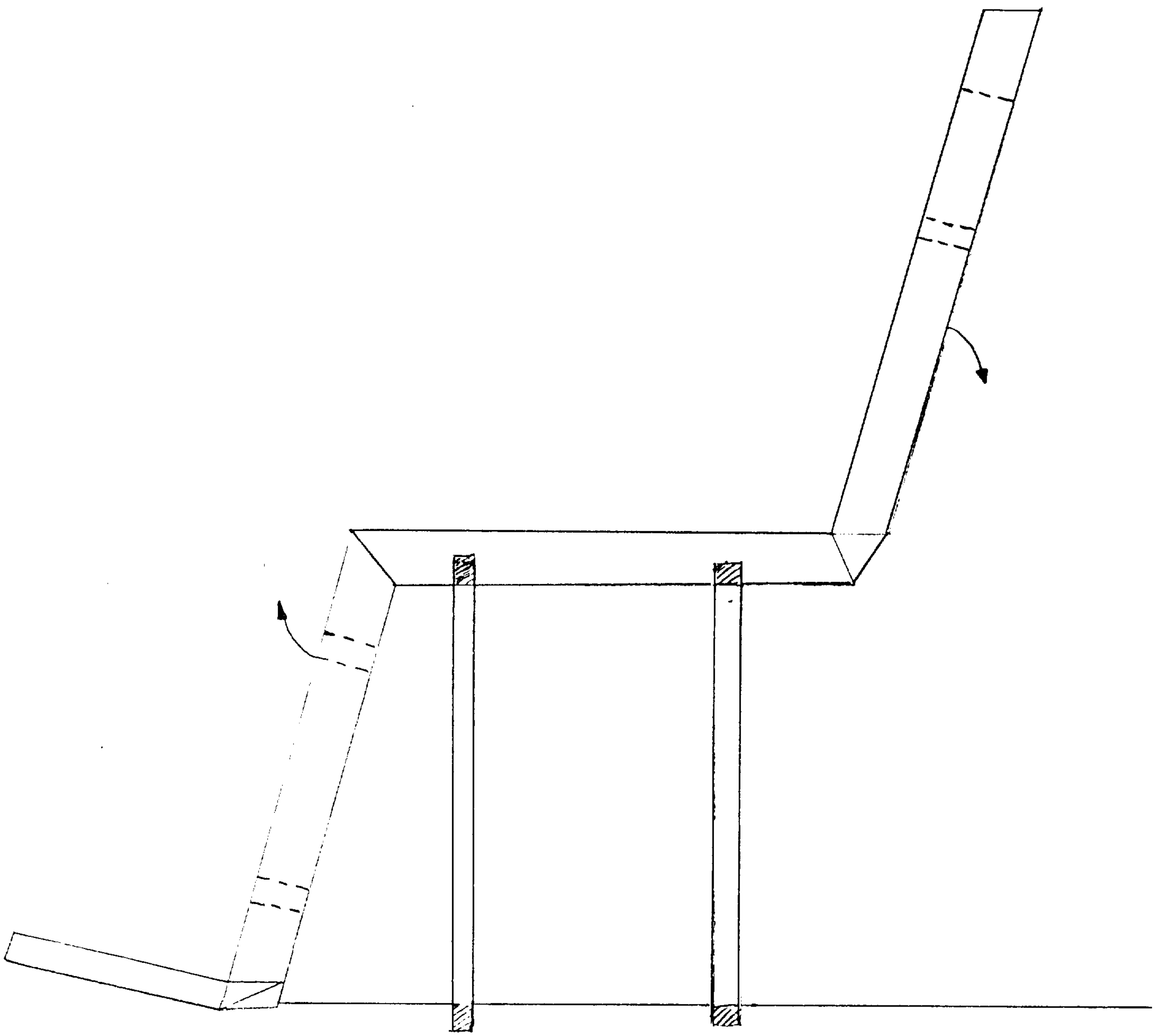
This chair can be quickly and easily set up by inserting the necessary support rods into their designated holes. In the event of an illness or injury , the chairs will be placed in the reclined position and then supported with the remaining support rods. These chairs will be sent up on one of the three initial launches. Once the chairs have arrived at the station they will be set up in each of the vehicles docked at that time. As new vehicles arrive at the station, the chairs will be dismantled for transfer to the new vehicle. The chairs will be returned to the station on the next available mission.

MISSION OUTLINE

The overall mission scenario is as follows:*

1.) Liftoff from Cape Canaveral
2.) Arrival at lift vehicle orbit(185.20 km)
3.) Begin Hohmann transfer to station orbit
4.)Arrive at station
5.)Unload cargo and move chairs to the new vehicle
6.)STINGRAE with most time at station begins reentry process
7.) STINGRAE lands at Cape Canaveral landing site**
8.)STINGRAE is returned to refurb. and proc. center
9.)STINGRAE begins payload integration

Logistics Chair



-m = 10Cm

10.)STINGRAE moved to launch pad to ready for launch

* The overall mission Delta V will vary depending on the station orbit
** In case STINGRAE is unable to land at Cape Canaveral its secondary landing site is Vandenberg Air Force Base. In case neither one of these landing sites is available, Reentry and Recovery has compiled a list of alternative landing sites.

PROGRAM IMPLEMENTATION

Projections estimate that the availability of the Titan IV launch vehicle with SRMU's to be 1994. However, design, development, and testing of STINGRAE will take in the neighborhood of five years. This means that the first phase of the seven year logistics resupply plan could begin as soon as 1995. The first launch will test STINGRAE's ability to maneuver into the proper orbits and then dock with the station. On the second mission, the chairs necessary for emergency situations will be taken to the station and remain on board until they can be distributed to the other vehicles. The third launch will lift the initial ninety day resupply requirements. New resupply missions will occur approximately every forty-five days and replace the craft which has been at the station the longest. Three vehicles will be in processing all of the time to make sure that a vehicle is ready for its scheduled launch. One vehicle however is set aside for testing.

TESTING

Testing of the STINGRAE system will continue throughout the program looking to always improve the performance and capabilities of

the STINGRAE. Various forms of testing are needed to make sure that the vehicle will be able to perform as designed and to find any flaws which could prove to be hazardous to the equipment or to human life. Testing is broken down into two categories: component testing and system. Component testing is used to make sure that each small part is working properly before it is integrated into the overall system. Once a part is accepted for overall system integration and assembled with all the other components system testing can begin. Some various components to be tested are: attitude and articulation control thrusters, main engine, computer systems, and communication system.

COSTING

One of the program requirements was to design a vehicle which is simple and low in cost. In designing the vehicle several components were used from already existing hardware in order to reduce design and development costs. The total system cost is based on a power curve cost estimating relationship. where:

$COST = A\ WGT_{SS}^{B}$ (source number)[1]

The total cost is then broken down into a cost for design, development, testing, and engineering (DDTE) and production cost (PROD). The equations used for these two individual costs are:

$COST_{DDTE} = A\ WGT_{SS}^{B} * (PND) * (DC) * (EI)$ [1]

$COST_{PROD} = A\ WGT_{SS}^{B} * (PC) * (EI) * (Quantity)$ [1]

where: PND = Percent New Design

DC = Design Complexity

EI = Escalation Index

PC = Production Complex

These formulas are given in terms of millions of dollars in 1978, and thus must be projected into1984 dollars which are then projected into 1989 dollars. The final costing analysis is as follows:

DDTE=843.306M PROD=117.218M TOTAL=960.524*M

* Cost given is for one vehicle

Interaction with other subsystems

In order to complete the task of coming up with a proposal for this program, communication with the other six subsystems was very important. Each subsystem needed some sort of information. Structures relied on the mass/volume requirements to approximate the size of the vehicle. Propulsion and Power needed to know what orbit the ELV would leave the vehicle at in order to calculate the amount of fuel needed. Life Support needed to know the crew size in order to determine the necessary supplies and tanks to provide. Attitude and Articulation needed to know how much maneuvering would need to be done. Command and Data Control needed to know a general mission plan in order to keep in touch with the vehicle. Reentry and Recovery needed to know where the main landing site was in order that other landing sites could be picked out in case of emergency.

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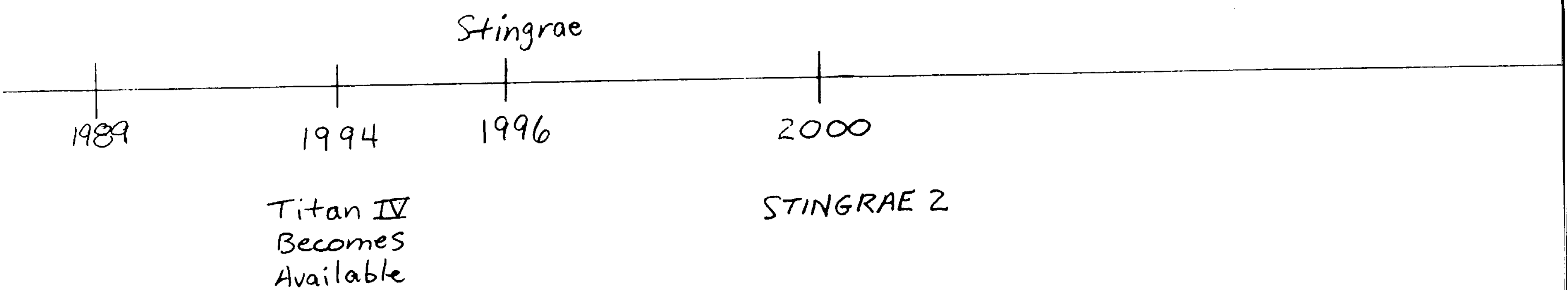
1 Lembeck,M.F.,AAE 241 Course Notes,Spring, 1989, Kinko's.

STINGLINE


Stingrae
1989
1994
1996
2000
Titan IV Becomes Available
STINGRAE 2

INTRODUCTION

Reentry and recovery is the final operational phase of the space mission. The problems associated with decelerating a reentry vehicle from hypersonic to subsonic speeds are complex. In addition to reentry concerns, the recovery of the spacecraft both in nominal and abort situations must be given full consideration.

Although the problem is complex, when broken down into its components it becomes more manageable. Reentry and recovery of the STINGRAE vehicle consists of the following parts: configuration analysis, trajectory analysis, thermal analysis and landing and recovery analysis. In general, the problem can be formulated as follows. Upon reentering the the earths' atmosphere with speeds between 20,000 and 50,000 feet per second, a reentry vehicle posesses an enormous amount of kinetic energy. Due to the density of the atmosphere, a substantial amount of drag reduces the velocity of the vehicles kinetic energy of motion and is translated into thermal energy. During reentry the vehicle absorbs some fraction of the total heat generated and creates unacceptable thermal loads for the crew and cargo inside of the vehicle. The solution is to reduce the heat absorbed by the vehicle. This can be accomplished in at least three ways: 1.) thermal protection systems 2.) spacecraft shape selection and 3.) reduce exposure time. Upon successful reentry into the earths' atmosphere, the vehicle must make either a land or water landing and then

retrieved so that it can be readied for another mission.

The STINGRAE vehicle was designed based on reentry and recovery requirements. The requirements were to:

a.) dissipate orbit energy in the atmosphere

b.) protect payload and crew from thermal and deceleration loads.

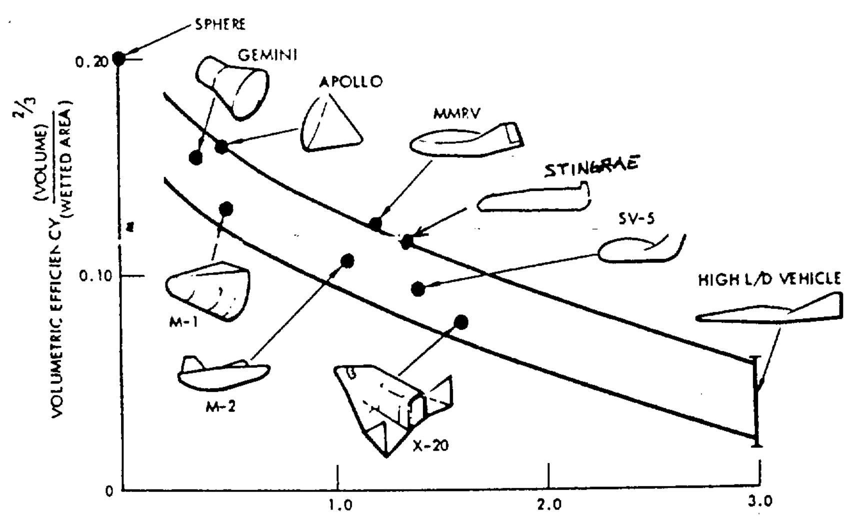
c.) carry out logistics module/crew pickup

CONFIGURATION ANALYSIS

STINGRAE vehicle configuration selection was critically driven by two factors: volumetric efficiency and payload fairing compatibility. The volumetric efficiency is governed by the following expression.

$$\frac{V^{2/3}}{S} = \text{volumetric efficiency}$$

The STINGRAE has a volumetric efficiency of .13 and can be compared with that of other vehicles in figure 1. below.



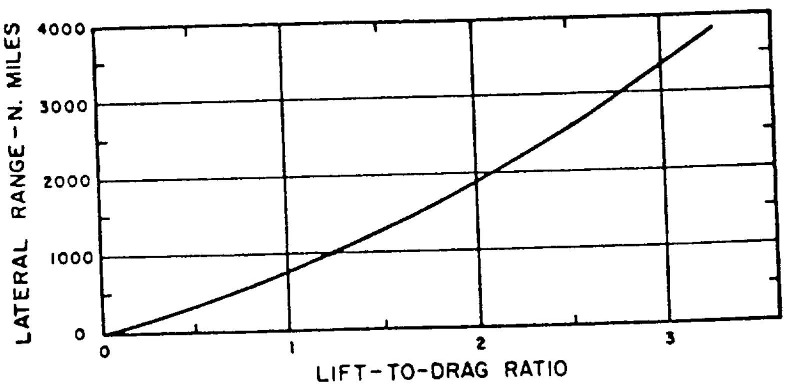
Typical configuration comparisons.

The Titan IV launnch vehicle used to insert the STINGRAE vehicle into low earth orbit (LEO) will encase the STINGRAE in its payload fairing. Therefore, considerable attention must be given to payload fairing size. With a payload fairing diameter of 16.67 ft. and a length of 56 ft., the STINGRAE vehicle is limited to a span of 15 ft. and a length of 54 ft. Based on this criterion, in addition to the need for precision and flexibility in landing, the choice was made to select a lifting body reentry vehicle configuration. The final STINGRAE configuration emerged only after several modifications were made to the original vehicle.

PERFORMANCE ANALYSIS

The aerodynamic characteristics of the STINGRAE vehicle must be calculated from experimental wind tunnel test data. The single most important aerodynamic variable is the lift coefficient which reflects the lifting capability of a particular surface at a given angle of attack. It is also a function of the shape of the lifting surface. Although wind tunnel tests were not conducted for this analysis, a preliminary value for the L/D ratio necessary to meet cross range specifications was obtained. Figure 2. yields a L/D ratio of approximately 1.3 required for a cross range manuevering capability of 1000 n.m. . A L/D ~ 1.3 places the STINGRAE vehicle in the medium L/D ratio category (.75 to 2.0) . Flight vehicle characteristics associated with the medium L/D category include: good weight, volumetric efficiency , and landing characteristics along with moderate range capabilities. The stability of the STINGRAE vehicle is dependent on aerodynamic variables such as the pitching moment coefficient and lateral static stability derivatives. Determination of

these values are obtained either through conventional wind tunnel studies or computational fluid dynamics programs.



Crossrange Maneuverability versus L/D

TRAJECTORY ANALYSIS

Protection of the crew and cargo from unacceptable deceleration or g loads was the primary design driver in this analysis. As outlined by RFP specifications, the maximum deceleration loads to be experienced by the vehicle is 3g's. Of secondary importance in trajectory analysis is obtaining values for entry velocity and deorbit delta-v so that apppropriate propulsion sizing may be determined.

The values for the entry velocity , entry angle, and maximum deceleration were obtained using an iterative process involving Homann transfer calculations. First, an intelligent guess for the (a) value was selected. This value was then used in eq.(4) until proper convergence occurred for the A_{max} value. Note that the A_{max} convergence value is 5g's and not 3g's because the higher L/D inherent in a lifting body type vehicle has the effect of flattening out the trajectory and thus reduces the maximum deceleration experienced by the vehicle. Assuming a flat

non-rotating earth and $C_D = 1$ the following relations were used to obtain entry angle, entry velocity and the maximum deceleration.

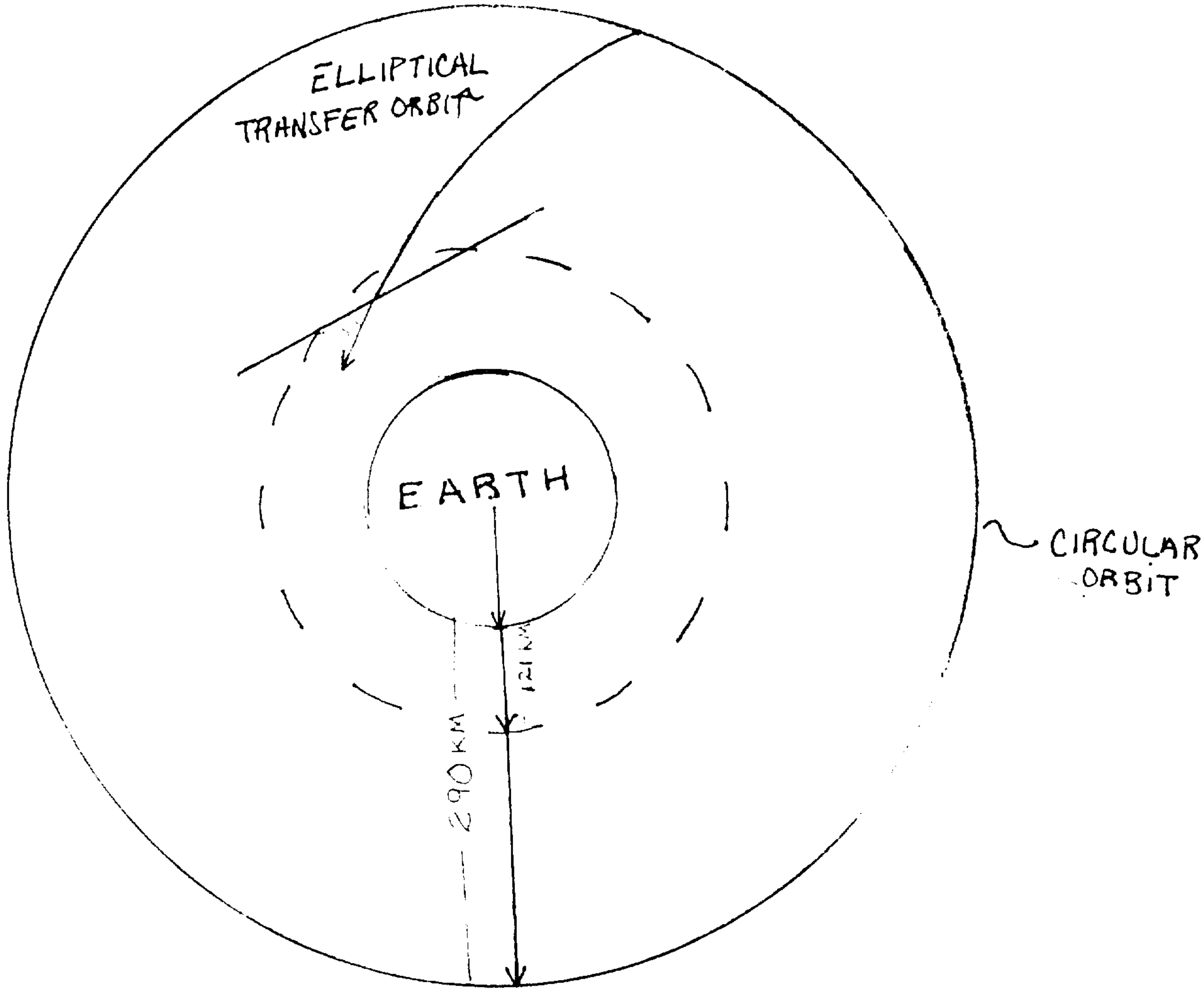
entry angle $$\gamma_E = \cos^{-1}\left[\frac{a^2(1-\varepsilon^2)}{r_e(2a-r_e)}\right]^{1/2}$$

entry velocity $$V_E = \mu\left(\frac{2}{r_e} - \frac{1}{a}\right)^{1/2}$$

max. deceleration $$A_{max} = \frac{V_E^2 \sin \gamma_E}{2 e g H_s}$$

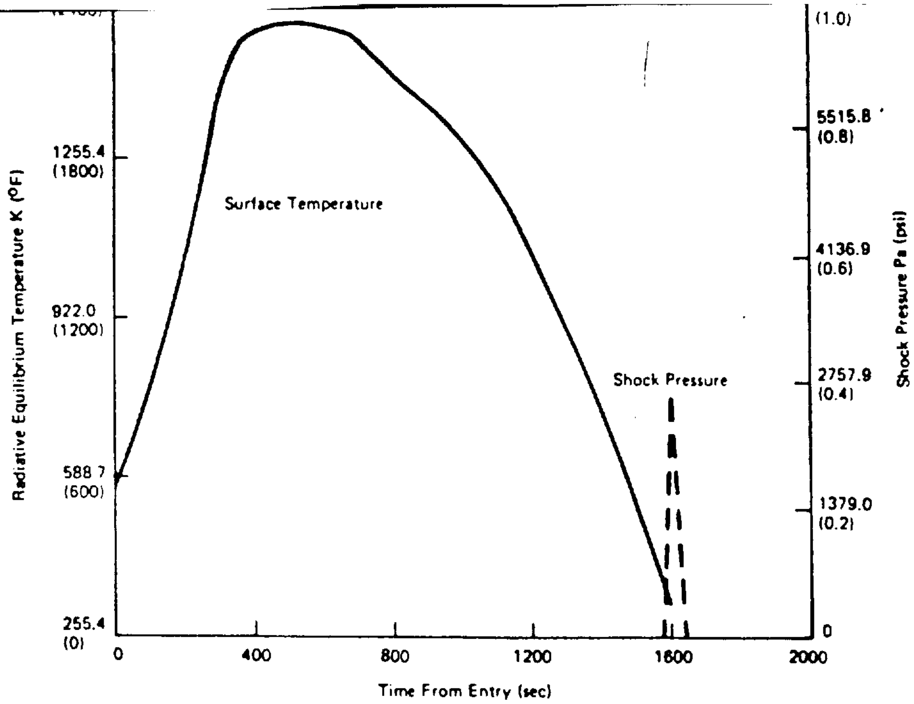
a (KM)	e	r_p (KM)	γ_E (DEG)	V_E (M/S)	A_{max} (G'S)
6500	.0258	6332	1.4813	7830.9	4.296
6450	.0338	6232	1.8855	7800.5	5.426
6460	.0332	6252	1.8107	7806.6	5.219
6465	.0314	6262	1.7724	7809.7	5.112
6470	.0306	6272	1.733	7812.7	5.003

TABLE .1 HOHMANN TRANSFER ITERATION RESULTS



TRAJECTORY ANALYSIS

ORBITAL ALTITUDE	290 KM
ORBITAL INCLINATION	28.5°
MAX. LATERAL RANGE	1000 NM
DEORBIT ΔV	.315 Km/s
ENTRY VELOCITY	7.812 KM/S
REENTRY ANGLE	1.73°
MAX. ENTRY ANGLE	6°
OPTIMAL ENTRY ANGLE	
DECELERATION LIMIT	3 g's



THERMAL ANALYSIS

The important design requirement addressed in this analysis is the protection of crew and payload from excessive thermal loads. One key design driver here is the maximum allowable temperature that humans can withstand. Consequently temperatures must not reach more than 150°F within the vehicle. Directly related to this constraint is the maintenance of 350 °F maximum temperature on primary structure imposed by the structures analyst. The lower the primary structure temperatures the less work required by the environmental control equipment.

During reentry the STINGRAE vehicle will experience two types of heat transfer phenomena: radiation and convection. Radiation occurs because of the thermally activated air molecules which have passed through the shock waves. Convection arises from the boundary layer of air flowing across the surface of the vehicle. To simplify the analysis of these heat transfer processes non-equilibrium effects and three dimensional effects were considered to be negligible.

The maximum external temperature experienced by the STINGRAE vehicle on reentry was determined to be 2400 °F. This value was selected upon analysis of figure 3. which plots the space shuttle temperature

profile as a function of time. Since the entry velocity of the STINGRAE vehicle (~ 7.81 km/s) differs from that of the space shuttle (~6.72km/s) by only 1km/s then similar amounts of heat are generated. The difference here being in the mass of the two vehicles. Although the masses differ causing the heat generated by the space shuttle to be higher, this design approximation has its merits(for conceptual design only !) in that it allows a rather large safety margin for temperature errors.

As stated previously in the introduction section, the motion of the vehicle and the thermal energy generated are directly related. When considering aerothermodynamics of STINGRAE the interdependence can be clearly seen.

Quantities of primary interest in this analysis include the peak stagnation heating rate ($\dot{q}_{max}$) and the total heat load (Q_0). The peak heating stagnation rate is the maximum heating rate occuring at the place where the fluid streamline is adiabatically decelerated to zero. The total heat load is of particular interest because it varies with the duration of heating. Exposure to a low total heat rates for long periods of time may absorb a larger total heat load than a vehicle with a high heat rate for a short period of time. It can be shown that exposure time in the atmosphere is directly proportional to the entry angle. The trade-off which yields optimal entry angle then is the intersection of the curves in figure 4. The expressions used to generate these values are:

Peak Stag. Heat Rate $$\dot{q}_{MAX.} = \frac{3.5 \times 10^7}{\sqrt{K R_n}} \left(\frac{V_E}{V_C} \right)^3 \quad \frac{W}{m^2}$$

Total Heat Load $$Q_0 = \frac{10^8}{V_c} \left(\frac{V_E}{V_C} \right)^2 \left\{ \frac{m}{C_D A \rho_0} \left(\frac{H \pi}{R_n \sin|\gamma_E|} \right) \right\}^{0.5} \quad \frac{J}{m^2}$$

The above quantities once determined can be used as input for thermal protection system (TPS) mass calculations. Thermal protection for STINGRAE was selected with the following criterion in mind.

1.) Light weight
2.) Effective isolation
3.) Durable for long service life and maintenance

The materials meeting the criterion and consequently selected are: reinforced carbon-carbon(RCC), titanium multiwall(TMW) , and carbon-carbon standoff (CCS).

Reinforced carbon-carbon will protect part of the nose of STINGRAE in addition to the leading edges of the wing and vertical stabilizer. RCC is a carbon cloth material immersed in a carbon rich matrix, heat treated , and coated with silicon carbide. The operational temperature range of RCC material is between (-250^{o}F to 3000^{o}F) .

Titanium multi-wall the titanium multi-wall panel is constructed of alternating layers of flat sheets of foil gage titanium and dimpled foil gage sheets, diffusion bonded to produce an integral prepackaged tiles complete with attachments. TMW use will be confined to the upperhalf of STINGRAE where surface temperatures are less than 1000^{o}F.

Carbon-Carbon Standoff has insulation which is secured between the vehicle skin and carbon-carbon panel. The heat shield is attached to

the vehicle using standoff supports. CCS material can withstand temperatures ranging from (2000°F - 2700°F).

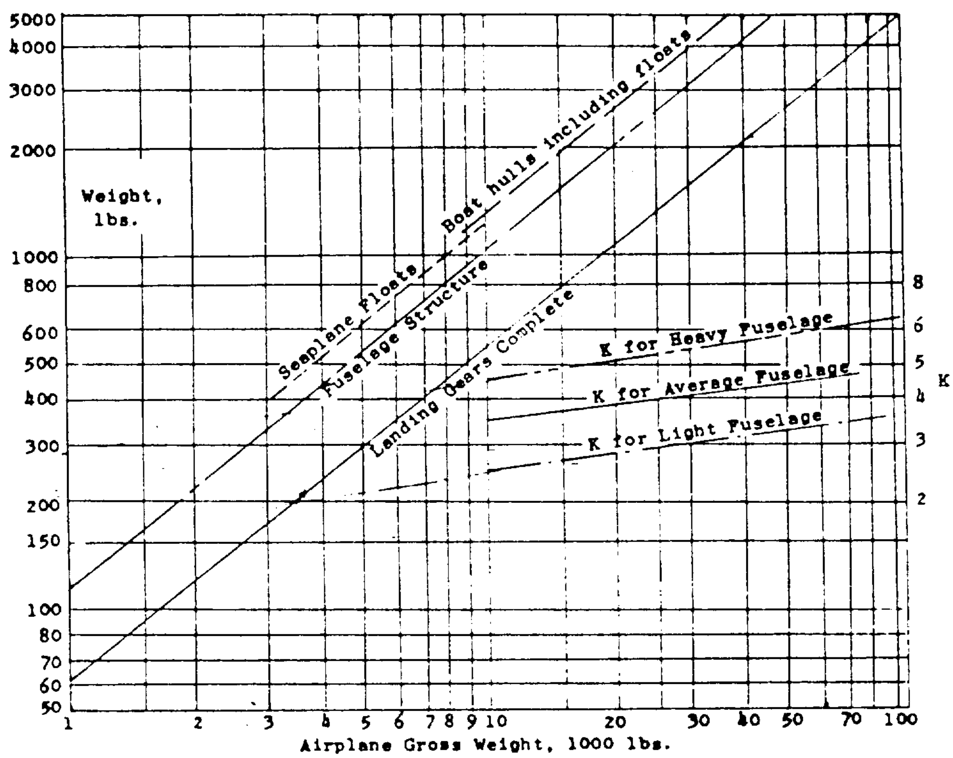
In order to obtain the TPS mass, the weight per unit area given in the panel specifications was multiplied by the surface area of the spacecraft over which the material was applied. These calculations were then handed over to the structures analyst to assist in pin-pointing the various mass contributions. It is worth noting here that an alternate method for calculating the TPS mass was explored but not used due to the lengtheness of the computations. The process made use of the linear conduction formula and the ITAS thermal analysis program in an iterative calculation for an arbitrary material selection.

LANDING AND RECOVERY ANALYSIS

After completing a successful reentry into the earth's atmosphere the STINGRAE vehicle will make a conventional aircraft landing tangent to the earths surface. A horizontal landing was selected for STINGRAE on the basis of the comparison study . The main advantage being the landing accuracy obtainable via this landing system. A typical landing and recovery scenario for the STINGRAE vehicle begins at deorbit and ends when the spacecraft and its occupants are safe on the ground. During this entire period , recovery personell will make appropriate recommendations concerning mission flight status and keep recovery forces informed of flight progress.

Once the vehicle is safely home it can be towed onboard any C-130 class cargo aircraft to its processing facility and once there will be refurbished and readied for its next mission. Safely home refers to those landing sites within the continental U.S. . Figure (7) gives the number of opportunities to reenter per fifteen full orbits as a function of both the orbit inclination and the required lateral range of the vehicle.

More specific landing analysis requires investigation into landing gear design. Tricycle landing gear were selected for the STINGRAE vehicle because of the advantages in approach stability, longitudinal trim, and improved ground handling capability. The type and size of the nose gear and main landing gear depends on the maximum static and dynamic loads placed on them. Using figure (8), the weight of the landing gear system can be determined. Also figure (9) lists the important values calculated from this analysis along with a diagram of the proposed landing gear configuration.



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AUTOMATED RESUPPLY CRAFT

- ARC -

Proposed design system in response to Request for Proposal for a Logistics Resupply and Emergency Crew Return System for Space Station Freedom.

AAE 241
May 2, 1989

Group 4

- ARC -

Group 4

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Table of Contents

ABSTRACT

This report describes the concepted design for the ARC or Automated Resupply Craft proposed to fulfill the requirements of the Request for Proposal for the Aerospace Vehicle Design Course, AAE 241. The ARC is designed to perform logistic resupply missions to Space Station Freedom. In addition, the design allows for the use of ARC as a crew emergency return capsule to bring astronauts back to earth from the space station. The ARC consists of three primary components: a logistics resupply module, space station docking adaptor, and an orbital transfer propulsion subsystem. The ARC's components and payload will be delivered to orbit on an expendable launch vehicle. The ARC is designed for a minimum six year lifetime, and uses technology available by 1994.

The following report is divided into the eight following subsystems: Mission Management, Environmental Control and Life Support, Command and Data Control, Reentry and Recovery, Structures, Attitude and Articulation Control, ARC Power and Propulsion, and the Orbital Transfer Propulsion Subsystem. Due to the loss of the group member responsible for structures midway through the course, the structure analysis and report is not as involved as the other subsystems.

Mission Management, Planning and Costing

Jim Bock

The mission management, planning, and costing (MMPC) considerations of the Automated Resupply Craft (ARC) serve to incorporate and integrate the remaining technical subsystems on the basis of several issues. These issues represent the specific requirements inherent to the MMPC subsystem in response to the submitted request for proposal (RFP) for a logistics resupply module and emergency crew return system, i.e., ARC, for Space Station Freedom.

Specifically, the MMPC requirements consist of the following: analysis and selection of crew options and vehicle number alternatives, identification and integration of required payload (for up and down missions), launch vehicle selecton, consideration of mission trajectory options, development of a mission planning timeline, identification of required ΔV for the missions, and analysis and estimation of mission costing requirements. The following is a study of the applicable options considered, the specific components selected, as well as the related technical rationale for each of the previously mentioned MMPC requirements.

First, an eight man crew is to be contained in a determined number of vehicles for each ARC system. Referring to Fig. 1, an analysis of the options relating to the crew size and number of vehicles is shown. From these choices, a system of two vehicles, each capable of accomadating four crew members, was chosen. This selection was based largely on optimizing logistics payload capacity as well as reducing unnecessary costs; for example, the one vehicle/eight man option was eliminated due to the resulting redundancy of consumable quantities and the obvious constraint of payload capacity, while the three and four vehicle alternatives imposed unfeasible costs to the system. As dictated in the RFP, a total of four of the two vehicle ARC systems will be constructed and implemented, with three systems being flight ready and the fourth system being retained for use in an integrated ground test system.

The required payload to be inserted in the ARC system accounts for the "logistics" term referred to in the RFP, and is comprised of various experiments, supplies, and/or waste to be taken to or returned from Space Station Freedom by the ARC system. In Fig. 2, the logistics totals derived for a ninety day period on Freedom are presented and categorized in terms of required mass and volume for both up and down ARC missions. The payload items will be neatly integrated, stored, and secured into an allowable portion of the ARC module, as revealed in the ARC structural layout.

The up missions of the ARC systems necessitate the utilization of a launch vehicle with the capability of allowing the ARC to gain access to Space Station Freedom. The Space Station access requirements as well as

those available launch vehicles that currently satisfy the requirements are listed in Fig. 3. The selection of the most compatible launch vehicle for the ARC system was based on a number of parameters. First, as shown in Fig. 4, a trade study of launch vehicle reliability versus cost was considered, with the reliability factor (success rate) carrying more influence than the cost in order to preserve the integrity and safety of the ARC missions. In addition, the analysis of launch vehicle payload capability versus cost (Fig. 5) served as a driving factor behind the selection of Titan IV upon consideration of the mass requirements of both the payload and ARC itself. Specifically, the mass and volume contributions of each ARC subsystem, listed in Fig. 6 and itemized by components in Fig.'s 7 and 8, represent a significant portion of the ARC systems total mass/volume requirements. These requirements, combined with the previously discussed payload mass and volume up mission contributions necessitated the division of up mission payload between two ARC vehicles in order to comply with the payload capacity of the Titan IV. The total ARC mass/volume requirements and capacities (with special consideration of the mass/volume capacities of the Titan IV) for both up and down missions of the ARC system are given in Fig. 9. Lastly, because the payload capability required of the Titan IV allows the ARC to gain an orbit of 100 nautical miles (refer to Fig. 5) while Space Station docking access dictates an orbit of 200 nautical miles (Fig. 3), an on-board chemical propulsion unit (integrated in the ARC system) will be utilized once the 100 nautical mile orbit is attained in order to reach the required 200 nautical mile Space Station orbit. For further explanation of the chemical propulsion subsystem, refer to that subsystem.

The trajectories of the up and down missions of the ARC system were designed in essentially two phases. The phases for the up missions consist of the trajectories from launch to Earth's atmosphere, and from the atmosphere to Space Station Freedom, while the two phases of the ARC down missions comprise the trajectories from Freedom docking to the Earth's atmosphere, and from the atmosphere to landing. The selected trajectory for the phase in which the atmosphere and the ground are the endpoints (up or down missions) is a ballistic path, while a Hohmann transfer trajectory will be implemented for the phase having the endpoints of the atmosphere and the Space Station (up or down missions). The chosen trajectories were largely based on the optimization of a number of technical issues, such as thermal shielding, g-forces experienced, and available working fuel. For a detailed analysis of the entry and reentry technical issues, as well as related justification of selected trajectories, refer to reentry and recovery subsystem.

The ARC system, as indicated in the RFP, is allotted a design lifetime of six years, which dictates a first launch occurrance in mid-1995 (assuming mid-1989 implementation of the system). With the ARC, the first three years of the design lifetime will be devoted to further conceptual design, technical study, and/or analytic research to

ensure the most feasible and efficient selection of components for the entire ARC-Space Station project. The remaining three years preceding launch will consist of the construction, installation, and testing of the ARC system and all of the technical subsystem components required. An outline of the final three year timeline in terms of major program milestones and integration of subsystem considerations preceding ARC's first launch at Cape Canaveral Air Force Station is given in Fig.'s 10 and 11. One remaining schedule to be considered is an outline of the ARC vehicle launch and return frequency. Referring to Fig. 12, two ARC vehicles are to be docked at Space Station Freedom at all times in the case of immediate, total crew return or an otherwise impulsive crew escape related emergency. This requires the employment of three nodes or docking facilities on the Space Station since, once two ARC vehicles are docked, a third ARC must dock Freedom to allow one of the two previously-docked ARC vehicles to return to Earth. The time related schedule of this cycle operates on a frequency of forty-five days; specifically, because the payload requirements (given in Fig. 2) for an up/down mission are designed for a ninety day duration, one division of the payload will be launched on one ARC vehicle on the first day while the remaining portion will be sent to the Space Station forty-five days later on a second ARC vehicle. Subsequently, with the duration of an up/down ARC mission of twenty-four hours (refer to the environmental control and life support subsystem for mission duration determination), a third ARC vehicle will be launched on the eighty-ninth day and dock on the ninetieth day to allow the first ARC vehicle to simoultaneously depart on the same ninetieth day-carrying the required ninety days of down payload. Again, this cycle is better understood with the aid of the ARC frequency timeline provided in Fig. 12.

The final requirements of the MMPC subsystem consist of the identification of required ΔV for the ARC missions and the analysis and estimation of mission costing values. A study of the determined mission ΔV and the related technical analysis for Earth-to-Space Station (and vice versa) maneuvers is referred to the advanced chemical propulsion subsystem , while the Space Station-to-orbital platform (and vice versa) ΔV analysis and requirements are referred to the electric propulsion subsystem. Lastly, a methodology of computing mission costs, and an estimation of the total ARC system costs are given in Fig. 13 in terms of each subsystem's contributing values as well as other related expenses. It is stressed that these figures represent only an estimation in the strictest sense, due to the potential exclusion of various program cost requirements as well as the possible over/under estimation of past, present, and future technology expenses.

In conclusion, the issues and requirements related to the MMPC subsystem of the ARC system in response to the submitted RFP for a logistics resupply module and emergency crew return system for Space Station Freedom have been presented with respect to the applicable options considered, the specific components selected, and the

related technical justification for each requirement. To reiterate, the particular MMPC requirements consist of the following: analysis and selection of crew options and vehicle number alternatives, identification and integration of required payload (for up and down missions), launch vehicle selection, consideration of mission trajectory options, development of a mission planning timeline, identification of required ΔV for the missions, and analysis and estimation of mission costing requirements. The MMPC subsystem serves to incorporate and integrate the remaining technical subsystems on the basis of the above requirements; what follows is the presentation of the technical studies, the analyses, and the conclusions exclusive to each of these subsystems.

REFERENCES

[1] Neilon, John J., Use of Expendable Launch Vehicles for OSSA Missions, Center for Space and Advanced Technology, Arlington, Virginia, 1988, pg. 18.

[2] Ibid., pgs. 6-23.

[3] Ibid., pgs. 29-53.

[4] "Space Station Cost Estimating Methodology for Hardware", Rockwell International (extracted from University of Illinois, Urbana-Champaign, AAE 241, packet #19, pgs. 13-14).

MMPC

OF VEHICLES vs. CREW SIZE

# OF VEHICLES	CREW SIZE (MAN/VEHICLE)
1	8
2	4,4
3	3,2,2
4	2,2,2,2

POINTS OF INTEREST:

- 1 VEHICLE OPTION LIMITS LOGISTICS PAYLOAD AND CREATES REDUNDANCY OF CONSUMABLE QUANTIES
- 4 ARC SYSTEMS COMPRISED OF 2 VEHICLES EACH WILL BE BUILT ; 3 FLIGHT-READY , 1 FOR INTEGRATED GROUND TESTING

FIGURE 1

MMPC

PAYLOAD ITEMIZATION

90 DAY TOTAL LOGISTICS REQUIREMENTS:

	PRESSURIZED (kg)		UNPRESSURIZED (kg)		FLUIDS (kg)		PROPELLANTS (kg)		MASS TOTALS (kg)	
	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER		
UP	4148.56	4954.14	513.01	4152.18	360.61	365.14	45.36	1681.92	16220.92	UP
DOWN	3497.99	4757.39	513.01	4152.18	0.00	173.73	0.00	0.00	13094.30	DOWN
	PRESSURIZED (m^3)		UNPRESSURIZED (m^3)		FLUIDS (m^3)		PROPELLANTS (m^3)		VOLUME TOTALS (m^3)	
	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER	CREW/STA.	CUSTOMER		
UP	14.78	13.92	4.53	32.64	0.45	0.50	0.57	1.68	69.06	UP
DOWN	11.50	13.75	4.53	32.64	0.00	0.17	0.00	0.00	62.59	DOWN

FIGURE 2

MMPC

LAUNCH VEHICLE REQUIREMENTS

- EXPENDABLE
- ABILITY TO LAUNCH PAYLOADS TO 220 NAUTICAL MILES
- INCLINATION LAUNCH OF 28.5° (AT CAPE CANAVERAL AIR FORCE STATION)

$\Downarrow$

THOSE THAT CURRENTLY QUALIFY:

- DELTA
- ATLAS-CENTAUR
- TITAN III
- TITAN IV (1994)

FIGURE 3 [1] (REF.)

MMPC

LAUNCH VEHICLE RELIABILITY vs. COST

LAUNCH VEHICLE	TOTAL # OF LAUNCHES	SUCCESSFUL LAUNCHES	SUCCESS RATE (%)	COST OF VEHICLE ($ MIL.)
DELTA	182	170	93.4	50
ATLAS - CENTAUR	66	56	84.8	59
TITAN III	135	130	96.3	110
TITAN IV *	—	—	"IN EXCESS OF 96.3"	125

* NOTE: TITAN IV IS SCHEDULED TO BE FULLY OPERABLE BY 1994

FIGURE 4 [2 (REF.)]

MMPC

LAUNCH VEHICLE PAYLOAD CAPABILITY vs. COST OF VEHICLE
(FOR AN ORBIT OF 100 N.M. X 100 N.M.)

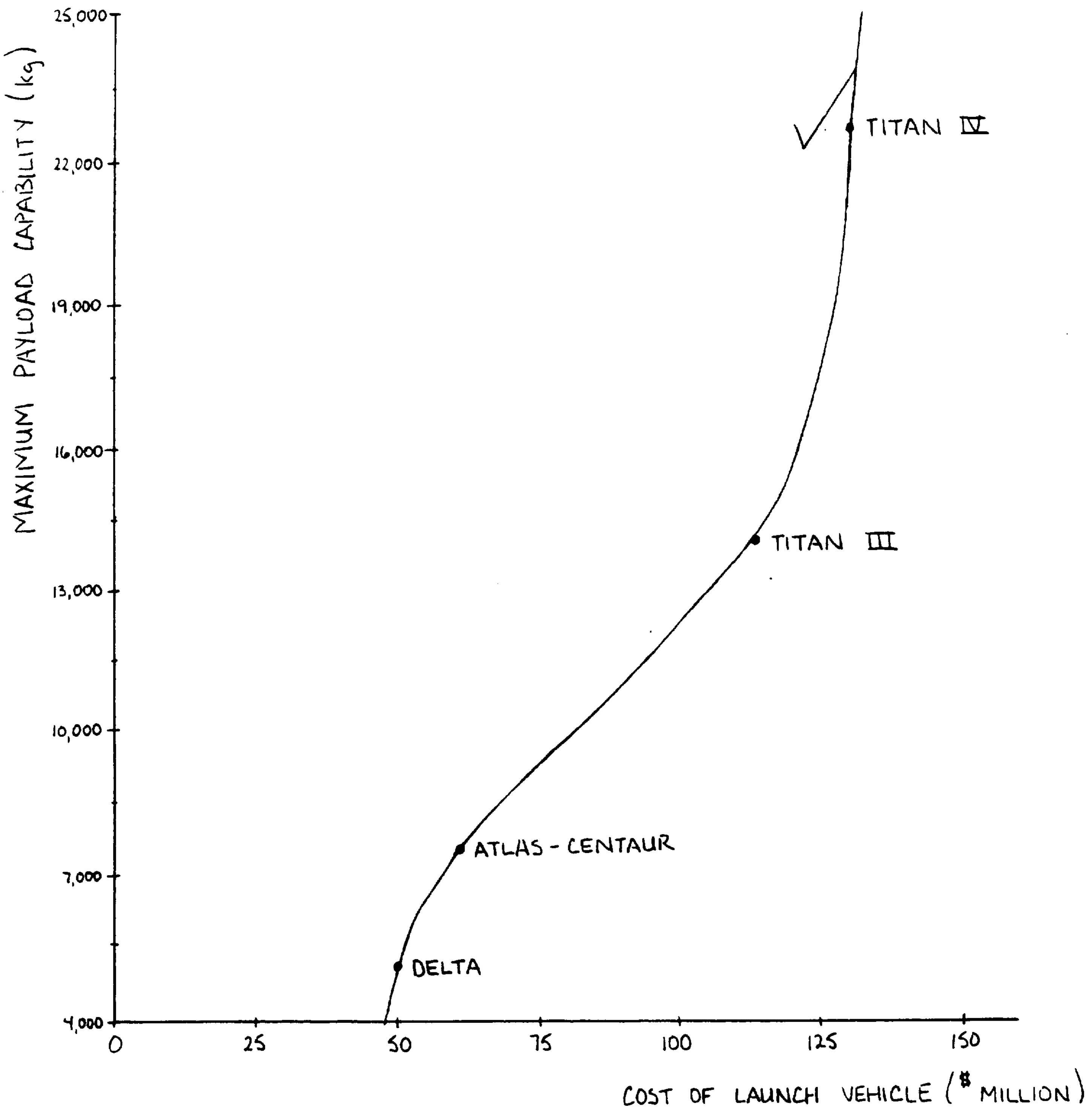


FIGURE 5 [3 (REF)]

MMPC

TOTAL SUBSYSTEM MASS / VOLUME CONTRIBUTION

	AACS	CDC	ECLSS	PPS	RRS	STRC	TOTALS
MASS UP (kg)	114.60	243.54	317.20	3772.88	1240.0	5028.02	10,716.24
MASS DOWN (kg)	114.60	243.54	607.50*	3220.08	1240.0	5028.02	10,453.74
VOLUME UP (m^3)	1.40	0.112	0.681	3.70	1.30	76.25** ([redacted])	7.193
VOLUME DOWN (m^3)	1.40	0.112	3.52*	3.70	1.30	72.62** ([redacted])	10.032

* THESE VALUES INCLUDE 4-MAN CREW VALUES (ALTHOUGH CREW WILL NOT RETURN WITH PAYLOAD)

** STRUCTURAL VOLUMES ARE SIMPLY PAYLOAD VOL. PLUS TOTAL SUBSYSTEM VOL.

FIGURE 6

MMPC

ITEMIZED SUBSYSTEM MASS/VOLUME VALUES

	UP MASS (kg/VEHICLE)	DOWN MASS (kg/VEHICLE)	
AACS	• TANKS = 33.6 kg • REACTION JETS = 0.10 kg • 3-SCAN C.M. GYROS = 80 kg • R - F.O. GYRO = 0.90 kg	SAME VALUES AS UP	UP TOTAL: 114.60 kg DOWN TOTAL: 114.601
CDC	• 4 ANTENNAS, 3 COMPUTERS = 173.54 kg • STAR, SUN SENSORS = 70.0 kg	SAME VALUES AS UP	UP TOTAL: 243.54 kg DOWN TOTAL: 243.54
ECLSS	• CONSUMABLES = 27.53 kg • GAS TANKS (O_2, N_2) = 29.52 kg • CABIN AIR SUBSYS. = 19.44 kg • THERMAL C'TRL. LOOP = 97.77 kg • PRESSURE C'TRL. SYS. = 1.36 kg • WATER STORAGE = 43.35 kg • INSTRUMENTATION = 9.412 kg • ATMOSPHERIC MASS = 88.82 kg	UP VALUES PLUS 4-MAN CREW MASS OF 290.3 kg	UP TOTAL: 317.20 kg DOWN TOTAL: 607.50
PPS	• ENGINE = 26.44 kg • HYDROZINE + TANK = 1388.8 kg • HYDROZINE AIR TANK = 818.2 kg • O_2 + TANK = 1001.6 kg • O_2 AIR TANK = 452.64 kg • SOLAR CELLS = (IN STRC) • BATTERIES = 85.2 kg	UP VALUES MINUS 552.8 kg BURNED FUEL	UP TOTAL: 3772.88 kg DOWN TOTAL: 3220.08
RRS	• REENTRY PARACHUTE = 1240.0 kg	SAME VALUES AS UP	UP TOTAL: 1240 kg DOWN TOTAL: 1240 kg
STRC	• STRC. COMPONENTS = 4048.5 kg • DOCKING ADAPTER = 313.0 kg • C-C BACK FLAP = 109.0 kg • SOLAR ARRAY ARM = 492.0 kg • HEAT SHIELD (NOSE) = 65.52 kg	SAME VALUES AS UP	UP TOTAL: 5028.02 kg DOWN TOTAL: 5028.02

FIGURE 7

MMPC

ITEMIZED SUBSYSTEM MASS/VOLUME VALUES (CONT'D)

	UP VOLUME (m^3/VEHICLE)	DOWN VOLUME (m^3/VEHICLE)	
AACS	• TANKS = .13 m^3 • REACTION JETS = 1.0 m^3 • 3-SCAN C.M. GYROS = .25 m^3 • R-F.O. GYROS = .012	SAME VALUES AS UP	UP TOTAL: 1.40 m^3 DOWN TOTAL: 1.40 m^3
CDC	• 4 ANNTENAS, 3 COMPUTERS = .1047 m^3 • STAR, SUN SENSORS = .0077 m^3		UP TOTAL: .1124 m^3 DOWN TOTAL: .1124 m
ECLSS	• CONSUMABLES = .00522 m^3 • GAS TANKS (O_2, N_2) = .0655 m^3 • CABIN AIR SUBSYS. = .0943 m^3 • THERMAL C'TR'L. LOOP = .166 m^3 • PRESSURE C'TR'L. LOOP = .0013 m^3 • WATER STORAGE = .159 m^3 • INSTRUMENTATION = .19 m^3 • ATMOSPHERIC MASS = —	UP VALUES PLUS 4-MAN CREW VOLUME OF 2.84 m^3	UP TOTAL: .6813 m^3 DOWN TOTAL: 3.52 m
PPS	• ENGINE = .1221 m^3 • HYDROZINE + TANK = 1.103 m^3 • HYDROZINE AIR TANK = .603 m^3 • O_2 + TANK = .720 m^3 • O_2 AIR TANK = .392 m^3 • SOLAR CELLS = .726 m^3 • BATTERIES = .032 m^3	SAME VALUES AS UP	UP TOTAL: 3.70 m^3 DOWN TOTAL: 3.70 m^3
RRS	• REENTRY PARACHUTE = 1.30 m^3	SAME VALUES AS UP	UP TOTAL: 1.30 m^3 DOWN TOTAL: 1.30 m
STRC	• ADD: • $V_{UP\ PAYLOAD}$ • + $V_{UP\ SUBSYSTEMS}$ • • = 76.25 m^3	ADD: $V_{DOWN\ PAYLOAD}$ + $V_{DOWN\ SUBSYSTEMS}$ = 72.62 m^3	UP TOTAL: 76.25 m^3 DOWN TOTAL: 72.62 m

FIGURE 8

MMPC

TOTAL ARC MASS / VOLUME LAYOUT

	PAYLOAD	SUBSYSTEM CONTRIBUTION	TOTAL	MAXIMUM CAPACITY
UP MASS (kg/VEHICLE)	8110.50	10,716.24	18,826.74	22,222
DOWN MASS (kg/VEHICLE)	13,094.30	10,453.74*	23,548.04*	26,937
UP VOLUME (m^3/VEHICLE)	45.50	7.193	52.69	84.00
DOWN VOLUME (m^3/VEHICLE)	62.59	10.032*	72.62*	84.00

* AGAIN, DOWN VALUES ACCOUNTING FOR 4-MAN CREW MASS / VOL. WILL NOT REALISTICALLY BE INCLUDED WITH PAYLOAD.

FIGURE 9

MMPC

THREE - YEAR MISSION TIMELINE

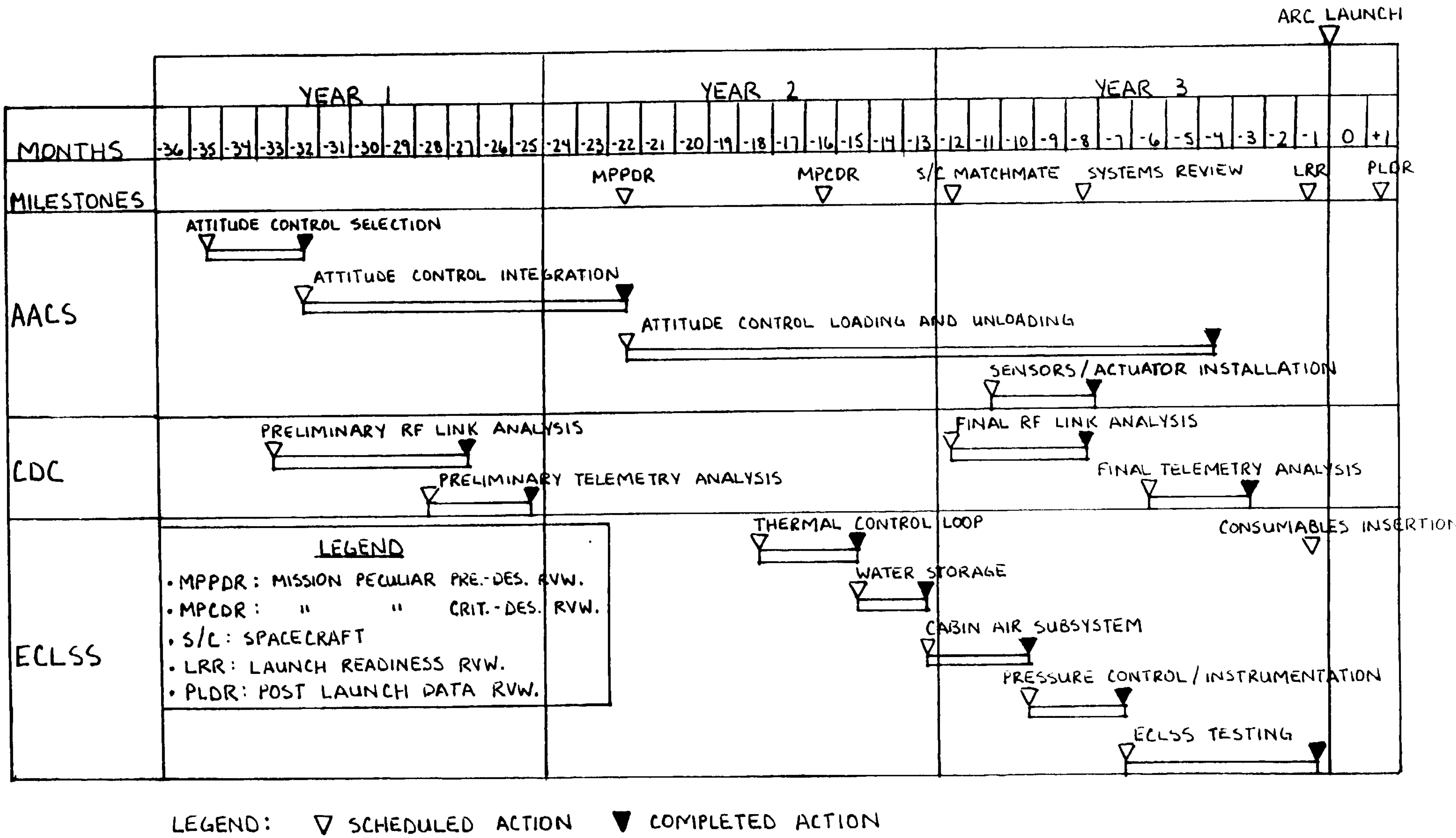


FIGURE 10

ARC- GROUP 4

MMPC

THREE - YEAR MISSION TIMELINE (CONT'D)

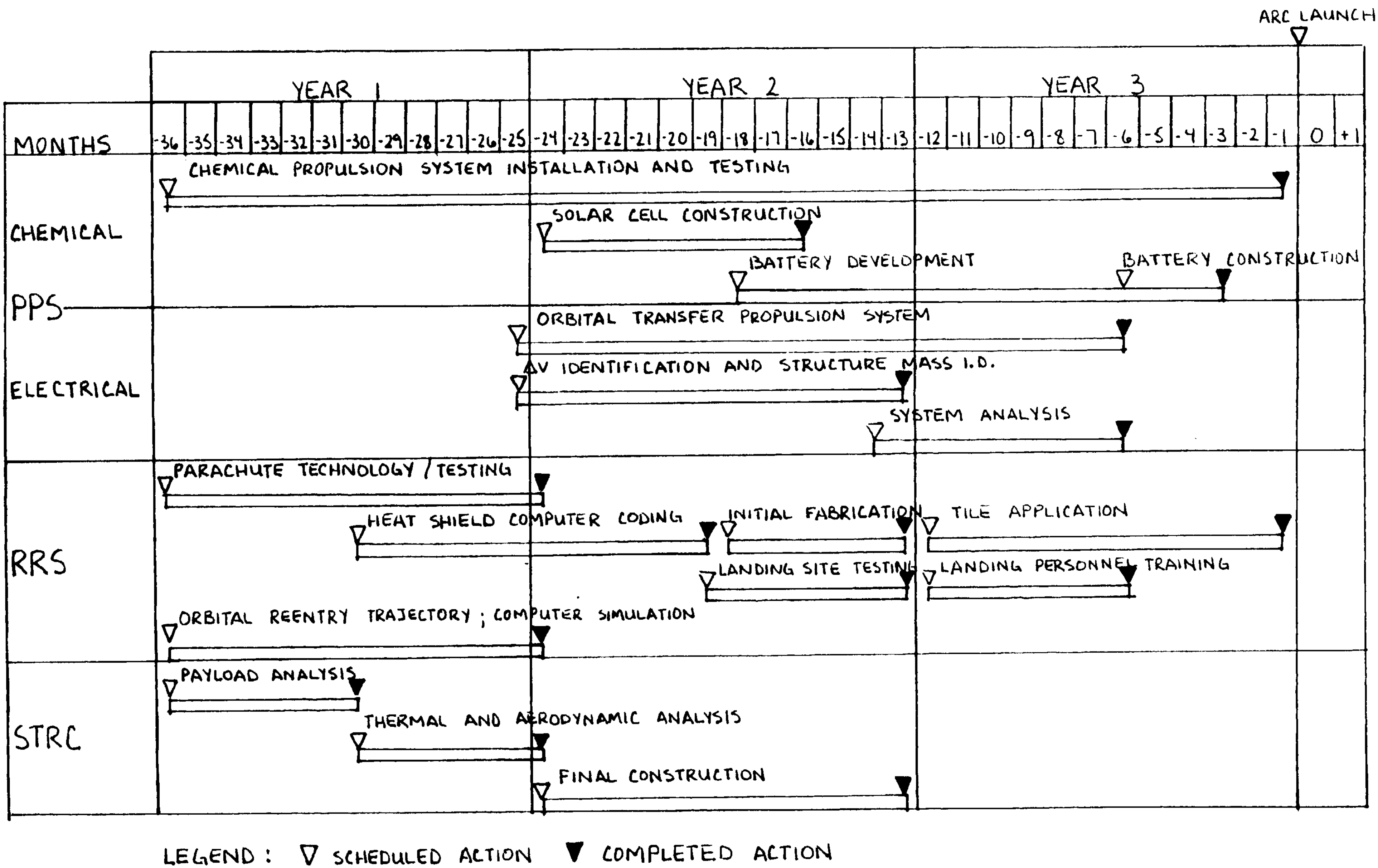


FIGURE 11

MMPC

ARC LAUNCH FREQUENCY SCHEDULE

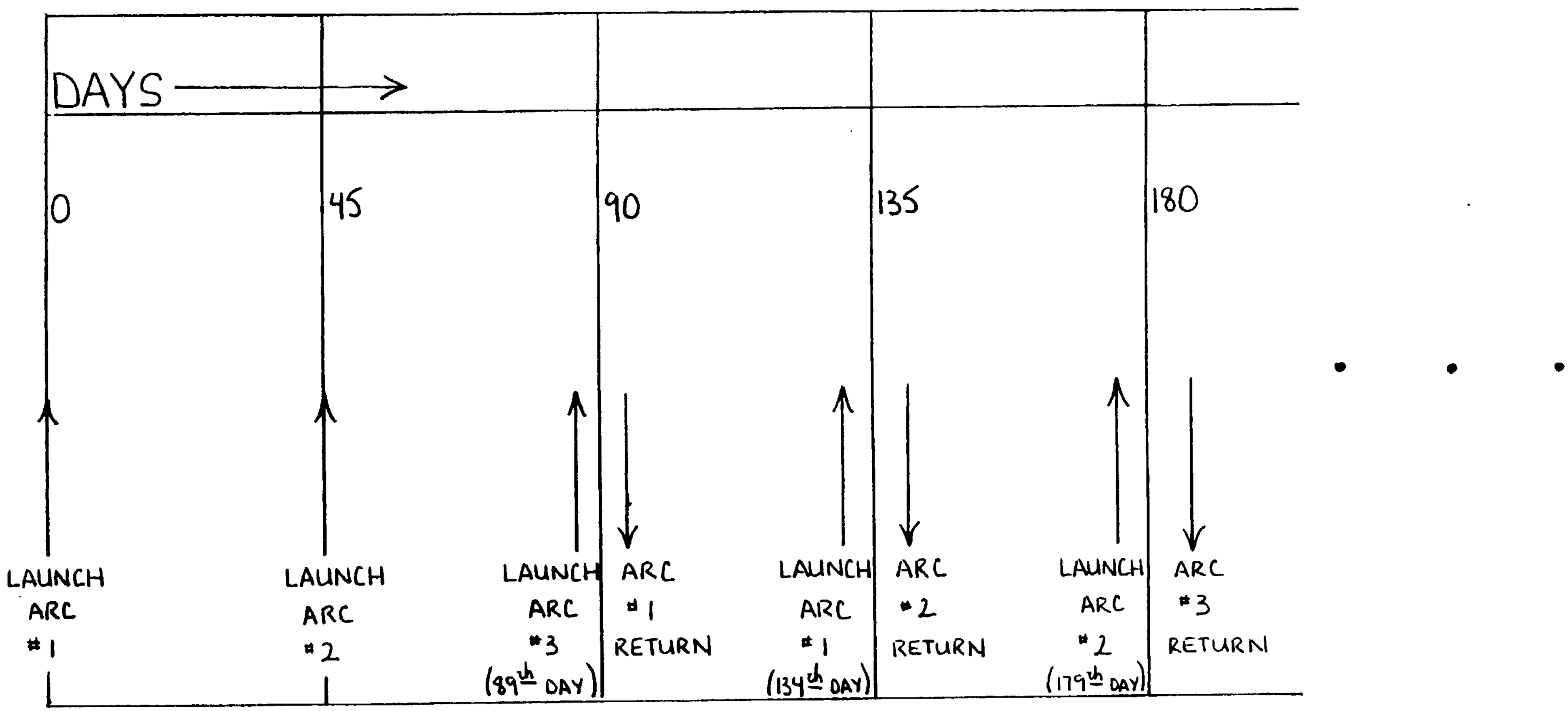


FIGURE 12

MMPC

MISSION COSTING ESTIMATES

	DDTE		PRODUCTION		COST
	COEFF.	EXP.	COEFF.	EXP.	($ MIL./VEHICLE)
AACS	4.57	.52	.86	.49	97.9
CDC	7.81	.58	.05	.92	308.4
ECLSS	5.86	.41	.40	.50	108.5
RRS	1.50	.26	.18	.55	29.1
PPS	.10	.88	.11	.55	235.8
STRC	1.76	.49	.42	.44	207.4
DOCKING	.45	.49	.06	.44	12.9
LAUNCH VEHICLE	—	—	—	—	125.0
				TOTAL =	$1.125 BILLION

- ARC SYSTEM TOTAL = 2($1.125) = $2.25 BIL.

- 4 ARC SYSTEM TOTAL = 4($2.25) = $9.0 BIL

COMPUTATIONAL METHODOLOGY:

- FORMULA USED*: $COST = C_{DDTE} + C_{PROD}$, WHERE $C_{DDTE} = A_{DDTE} W^{B_{DDTE}}$, $C_{PROD} = A_{PROD} W^{B_{PROD}}$
  AND $A \Rightarrow$ COEFF., $B \Rightarrow$ EXP., $W \Rightarrow$ WEIGHT

* NOTE: ACTUAL VALUES FROM EQUATIONS DO NOT REFLECT TABULAR VALUES; THESE EQUATIONS WERE USED FOR SUBSYSTEM SCALING PERCENTAGES ONLY.

FIGURE 13

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM

Michael J. LeDocq

The logistics resupply vehicle (LRV) must also function as a crew emergency return vehicle (CERV). Therefore, an environmental control and life support subsystem must be designed which will support space station crew members during an evacuation. It was determined that the CERV should be able to support up to four crew members for a maximum of 24 hours. Because the size and duration of the rescue mission are small, a regenerative ECLSS would be too large and complex. Non-regenerative ECLSS using expendable supply of consumables will be used. O_2, N_2, and potable H_2O will be supplied in tanks without recovery. LiOH will be used for CO_2 removal. A two-phase thermal control system will be used as well as existing fire suppression and smoke removal equipment.

CREW SIZE VS. LIFE SUPPORT REQUIREMENTS

The relationships of consumables, metabolic heat production, and volume requirements are all linearly related to crew size. The most important design factor seems to be the mass of the O_2/N_2 tanks which become very massive as the crew size increases. It was determined with the mission planning analyst that in order to design for sufficient volume for cargo, the crew size should be as small as possible, limiting the size of the ECLS subsystem. It was also determined that the LRV system of two vehicles should be capable of evacuating the entire crew of eight astronauts. It was decided that each vehicle should be capable of carrying up to four crew members for a maximum of twenty-four hours.

SIZING OF CONSUMABLES

Crew rescue missions during which the ECLSS will be used is a secondary requirement of the LRV and these missions will be no longer than twenty-four hours. Regenerative ECLS subsystems such as revitalization of cabin air and waste water reclamation would introduce unnecessary complexity, mass, and volume to the LRV. For a mission this short, expendable (open-loop) methods of supplying consumables are not prohibitively large. For

these reasons, and open-loop ECLS subsystem will be used. While docked in orbit, the LRV will utilize the air revitalization and thermal control systems of the space station in order to maintain a habitable atmosphere in the vehicle at all times. These subsystems will be connected to the space station through interfaces in the docking ports of the LRV and space station.

O_2, N_2 TANKS and CO_2 REMOVAL

Consumable O_2 and replacement N_2 can be stored in pressure vessels or in cryogenic vessels. A chart showing the advantages and disadvantages of these methods is shown in Figure 1. Pressure vessels were chosen primarily because of their longer shelf-life. This is needed because the vehicle will be docked at the space station for 90 days, during which the gas supplies must remain intact and useable. The cryogenic tanks also require cooling equipment which would add to the size and power requirements of the ECLS subsystem.

The O_2 required to support a four person crew for twenty-four hours was found to be 8.32 lbm or 3.78 kg, and the mass of N_2 required to

	Pressure vessels	Cryogenic tanks
Advantages	Small space Ambient temperature Long shelf life	Small space Light-weight Thin walls
Disadvantages	Heavy Possible explosion Leakage	Cooling & insulation Short shelf-life

Figure 1. Pressure vessel vs. cryogenic storage comparison.

replace atmosphere leakage over this time period was calculated as 3.60 kg. In order to size the tanks required to store these gases, the thin-wall pressure vessel analysis was used (see Appendix). The gases were assumed to be stored at 3000 psi, 80 °F, with a safety factor of 3. The material used was stainless steel. The calculated tank sizes are shown in Figure 2.

These tank masses appear unreasonably large. It can also be seen that the thicknesses of both tank walls are approximately 1/4-th to 1/3-rd of the radius of each tank. These facts indicate that the thin-wall analysis is

	Volume (m^2)	Mass (kg)	Wall Thickness (m)	Outside radius (m)
O_2	0.0445	233.3	0.063	0.224
N_2	0.0505	274.3	0.058	0.233

Figure 2. Thin-wall pressure vessel sizes.

inadequate for these tanks. The total mass of high pressure storage tanks with stored gas has been shown to be approximately four times the mass of the stored gas (Heitchue, 1968, p. 174). This yields gas plus tank masses of 15.12 kg and 14.4 kg for the O_2 and N_2 tanks respectively. These masses will be used for subsystem sizing (see Figure 4). The previously found volumes will be used as approximations to ensure enough space for these tanks in the vehicle.

Expendable LiOH will be used to remove CO_2 from the cabin atmosphere during crew evacuation. The maximum required LiOH for this mission is 5.45 kg which occupies 0.00374 m^3. The LiOH will remove the necessary amount of CO_2 from the atmosphere and the used LiOH can be removed and replaced during ground servicing. The CO_2 removal system will remain inactive during routine resupply missions.

POTABLE H_20 and SOLID FOOD

The maximum required mass of potable H_2O for this vehicle is 14.70 kg and will occupy 0.148 m^3. The water supply system will utilize existing technology.

Most evacuation missions during which the ECLSS will be used will last no more than a few hours. Because of this, no solid food will be stored on the LRV. If necessary, food items can be taken aboard the LRV from the space station supplies as needed for the crew members leaving the space station. The mass and volume of these foodstuffs would be negligible.

REQUIRED CREW VOLUME

The equations used to determine the minimum and acceptable volume required for crew members are included in the Appendix. These equations vary quadratically with respect to mission length, making a greater volume of

open living space a limiting factor as mission length increases. But, if mission length is predetermined, the volume requirements become linearly dependent on crew size, as are the rest of the required consumables. The acceptable volume required for four crew members for twenty-four hours is 9.78 m^3, much less than the 69.06 m^3 required for resupply cargo (AAE 241, Feb 23,1989). The volume required for all eight crew members is only 19.6 m^3, again much less than the available space in an empty LRV. The open space volume requirement for up to four crew members is easily met with and empty LRV used for evacuation purposes, but cannot be met in an LRV on a routine resupply return to earth when filled with waste cargo. In the case of a minor injury, e.g. a broken leg, which may not require immediate evacuation, but a return to a 1-g environment within a short period of time, some cargo may be stowed in the LRM with the crew member(s) if necessary or feasible.

CABIN ATMOSPHERE

A cabin atmosphere which is compatible with the space station atmosphere will be used to pressurize the LRM. This atmosphere has a pressure of 14.7 psia with 21% oxygen content and nitrogen as a diluent. This atmosphere will be used to pressurize the LRM for routine resupply missions as well as during crew rescue missions. The mass of this O_2/N_2 atmosphere at 14.7 psia and 75 °F which fills 75 m^3 is 88.82 kg. This value is included in the mass totals for the LRV, and does not include CO_2 or H_2O vapor.

In order to maintain an acceptable invironment for crew members, the cabin humidity and temperature must be controlled. It is desirable to use and active thermal control system, since a passive system is controlled only by the amount of heat generated inside the spacecraft (Heitchue, 1968, p. 196). The most effective active thermal control method uses radiation to space for heat dissipation (Heitchue, 1968, p. 196). The systems included in spacecraft have most often used a heat pipe system with a circulating single-phase fluid to transfer heat and a radiator (e.g. radiator fin or coldplate) to dissipate heat to space. Another form of active thermal comtrol uses a two-phase fluid to transport and dissipate heat. Prototypes of two-phase systems have been tested during the 1980's, and useable systems should be producible by 1994.

Heat pipe thermal control systems are reliable, have simple designs, and have high thermal efficiencies. These systems usually require cryogenic cooling, which adds weight to the subsystem (Groll, 1987). The use of

radiator fins is another drawback because of size limitations on this vehicle and their inability to be used during reentry.

Two-phase systems have a small mass and require only a small pumping power. Cold plates or radiators in a two-phase system operate at a nearly constant temperature, which can be near the ambient temperature of the cabin atmosphere (Grote, 1987). These systems use fluids such as ammonia or freon, which could contaminate the atmosphere if a leak occurred. A two-phase system with cold plate can dissipate up to1 kW of energy. Some systems can operate either actively or passively by using capillary forces for circulation and mechanical pumping to start/restart circulation, or for higher rates of heat dissipation (Kreeb, 1987). Radiator fins are also capable of dissipating large amounts of heat (> 1 kW), but can be as large as three meters long (Tanzer, 1988) and are better suited for larger spacecraft. It was determined that a two-phase thermal control system would be used because this system is less massive, requires less power, and because cold plates which are capable of dissipating more than the required 625 W (see Figure 4) can be developed.

	Advantages	Disadvantages
Heat pipe w/ or w/o radiator fin	Reliable Simple design High thermal efficency	Need cryogenic cooling Radiator fin(s) no reentry use large radiator length
Two-phase	Small mass Small pumping power Constant temp. operation active or passive	use ammonia or freon

Figure 3. Comparison of thermal control systems.

Humidity control will utilize existing technology and will be incorporated into the thermal control subsystem where water vapor can be condensed and collected.

FIRE DETECTION and SUPPRESSION

Fire detection and suppression can be carried out by using smoke and heat detectors and fire suppression techniques which are currently available. This system should be automatic because of the absence of a crew or the

limited mobility of any returning astronauts. A smoke removal device which is capable of removing smoke particles and toxic gases produced by a fire could be included in the system. A smoke removal unit would be useful in the LRV if a fire in the space station causes the crew to use the vehicle for evacuation. Any contaminants in the vehicle atmosphere could be removed independently from the space station circulation system. A smoke removal unit prototype which contains filters for smoke and toxic gases has been designed and tested for use aboard Navy ships (Birbara, 1988), and if it is successful, a similar device could be designed for use aboard the LRV. The prototype has dimensions of 23"W x 27"D x 72"H and would require a relatively small volume.

THREATS

The reason for providing this resupply vehicle with a life support subsystem is so that it can be used as a crew emergency return vehicle (CERV) in the event that one or more crew members must be evacuated from the space station. A situation which puts the crew or the spacecraft in danger is defined as a threat. Some major threats which could cause space station evacuation are fire, biological or toxic contamination, injury/illness of crew member(s), explosion/implosion, loss of pressurization, radiation, meteriod/debris collision or penetration, stores/consumables depletion, tumbling/loss of control, orbit decay, and out-of-control extravehicular astronaut. The threat of fire and contamination has been dealt with by including fire suppression and smoke removal equipment in the LRV. Loss of pressurization, tumbling, and orbit decay can be caused by meteoroid penetration, collision, loss of power or fuel, explosion of a pressure tank or thruster failure. If the failure cannot be repaired in a short amount of time, the space station might have to be evacuated. An out of control astronaut could be caused by the above conditions or astronaut illness. It could be possible to configure the LRV to act as a rescue vehicle in this event. The logistics vehicle system can evacuate part or all of the space station crew members. A partial evacuation could be caused by an injury or illness or an out of control astronaut. All other threats would most likely warrant a complete evacuation (AAE 241, Feb 2, 1989).

MEDICAL CONSIDERATIONS

The seats of the LRV should be designed to fold down flat and lock in place in order to serve as beds or

supports on which stretchers could be placed in the case of an injury/illness where a crew member(s) must remain immobile. A winch or other lifting device should be mounted near the hatch in order to lower and raise the stretcher(s) in and out of the spacecraft. When no crew members are present in the vehicle, the seats can be folded down to increase space for stowage of cargo. The reentry analyst would design trajectories which limit g-forces to acceptable human limits, but the seats can be equipped with a shock absorbing apparatus. Medical supplies and first aid treatment should be provided on the space station, thereby eliminating the need for medical supplies on the resupply vehicle.

INTERACTION WITH OTHER SUBSYSTEMS

The ECLSS subsystem interacts with several of the other subsystems, as shown throughout this report. The most important interaction is with the mission planning subsystem to determine the mission length and acceptable number of crew members to be designed for. Interaction also occurs with the structures subsystem to ensure that the spacecraft walls have been designed to protect the crew from radiation (thick wall) and from micrometeoroid penetration and spalling (inner wall). The reentry subsystem must ensure the use of trajectories which limit reentry g-forces to safe levels. The power and propulsion subsystem provides power to most of the ECLSS components.

ECLSS SIZE TOTALS

The following summary of the mass, volume, and power requirements for the environmental control and life support subsystem. Sizes of the major components of the ECLS subsystem were scaled down, where necessary, from figures provided for a crew rescue vehicle capable of supporting six crew members for thirty-six hours (AAE 241, Feb. 7, 1989).

	Mass (kg)	Volume (m^3)
O_2 use:	3.78	0.0143
N_2 leakage:	3.60	0.0155
LiOH use:	5.45	0.00374
H_2O produced:	9.98 (to be disposed)	0.010
H_2O consumed:	14.70 (to be supplied)	0.0148

Figure 4. ECLSS mass, volume, power totals.

consumables totals:		
up	27.53	0.01854 (does not include O_2,N_2)
down	37.51	0.02854
gas tanks:		
O_2:	15.12	0.0301
N_2:	14.40	0.0354
ECLSS components:		
Cabin air subsys.:	19.44	0.0943
Thermal control loop:	97.77	0.166
Pressure control subsystem:	1.361	0.0013
Fluid storage (thermal control):	43.35	0.159
Instrumentation:	9.412	0.19
Cabin atmosphere:	88.82	75.0 (inside vol. of LRV)
TOTALS:		
Up:	310.183	0.6813
Down:	600.5 (includes 4 persons at 72.6 kg each)	0.6813

Metabolic heat: 625.4 W

V_{min}: 9.38 m^3

V_{acc}: 9.78 m^3

Power consumption: 854.2 W (max) 293.4 W (ave.)

Figure 4. ECLSS mass, volume, power totals (cont'd).

TECHNICAL PROBLEMS

There should be few technical problems in the development of the ECLS subsystem because all of the non-regenerative supply processes have been used in spacecraft. New technology includes the two-phase thermal control system and the compact smoke removal device. Prototypes for both of these types of systems have been built and tested, but they have not been produced for actual use in a vehicle. The two-phase thermal system is being developed for dissipation of heat loads much greater than is generated by this vehicle. Also, the prototype and testing phase just described for both systems was reached two to three years ago, so successful, full-scale

models may be near production now. Even if these systems cannot be developed by 1994, an existing single-phase thermal system can be used, and the smoke removal unit can be omitted from the system without endangering crew members.

CONCLUSION

Because the support of a crew during a space station evacuation is a secondary function for the logistics resupply vehicle, and because of the short duration of such a mission, a non-regenerative life support system is used. Oxygen and nitrogen for the cabin atmosphere will be stored and supplied from high pressure tanks. LiOH will be used to remove CO_2 from the cabin, water will be provided without reclamation and no food will be provided. A two-phase thermal control subsystem with a cold plate heat dissipater will be used for temperature control. A fire detection and suppression subsystem utilizing current technology is used, as well as a smoke removal device. Medical supplies will not be supplied, but immobile crew members can be transported in a prone position, either secured into an extended seat or in a stretcher which is secured in the vehicle. Each vehicle is designed to support four crew members for twenty-four hours; a logistics supply system consisting of two vehicles is capable of evacuating the entire space station crew of eight if necessary.

APPENDIX

ECLSS Requirements

O_2 use:	2.08 lbm/man-day
N_2 leakage:	0.33 lbm/h
LiOH use:	3.0 lbm/man-day
H_2O exhaled:	5.5 lbm/man-day
Metabolic Heat:	533 Btu/man-hr

(AAE 241, Feb. 7,1989)

Stainless steel properties:

$\rho = 0.28$ lbm/in^3
$S_y = 30{,}000$ psi

Gas Constants:

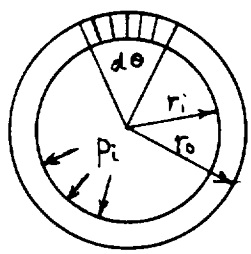
$R_{O2} = 48.28$ ft-lbf/lbm-°R
$R_{N2} = 55.15$ ft-lbf/lbm-°R

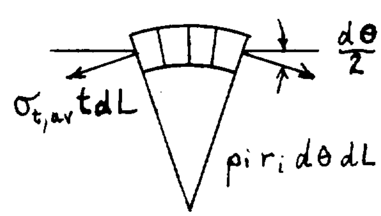
Potable H_2O (lbm/man-day):	6.8-8.1
Cabin Temperature (°F):	65-75
Relative humidity (%):	25-75
O_2 partial pressure (psia):	2.85-3.35
CO_2 partial pressure (mm Hg):	3.0 max

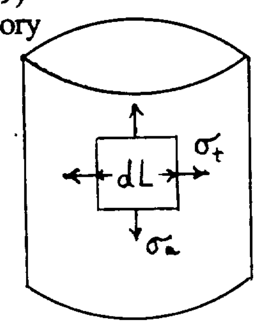
(Miller, 1987, p. 198)

$$V_{min} = -(0.0040)\,x^2 + (1.4219)\,x + 81.307 \text{ ft}^3 / \text{man-day} \quad (x = \text{days})$$
$$V_{acc} = -(0.0068)\,x^2 + (2.8346)\,x + 83.440 \text{ ft}^3 / \text{man-day} \quad (x = \text{days})$$

Pressure Vessel Sizing (AAE 241, Feb 7, 1989)
Internally Pressurized Cylinder - Thin Wall Theory







Equilibrium of forces acting radially on element yields:

$$p_i\, r_i\, d\theta\, dL = 2\, \sigma_{t,av}\, t\ dL\, d\theta/2$$

$\sigma_{t,av} = p_i\, r_i / t$ - average tangential stress $(t << r_i)$

$\sigma_{t,max} = p_i\, r_{av} / t$ - maximum tangential stress, where $r_{av} = r_i + t/2$

If ends of the cross-section are closed, axial force of magnitude $p_i\, \pi\, r_i^2$ is

distributed over cross-sectional area, $\pi\,(r_o^2 - r_i^2)$. This yields

$A = 2 \pi r_{av} t$, and

$\sigma_{a,av} = p_i r_i^2 / (r_o^2 - r_i^2)$ - average axial stress

$\sigma_{a,max} = p_i r_i^2 / (2r_{av}t)$ - maximum axial stress

To size tanks:

- compute required volume from ideal gas law $PV = mRT$

- compute r_i of tank from $V = 2 \pi r_i^3$

- compute axial and tangential stresses - $\sigma_{t,max}$ will always be largest for this theory

- take (largest stress) * (safety factor) = (yield strength)

- compute thickness of wall

  $= S_y / (\text{safety factor}) = p_i r_{av} / t = p_i / t + p_i / 2$

- compute mass of tank

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COMMAND AND DATA CONTROL

Ronald Gliane

Introduction

This subsystem has four main requirements as designated in the RFP: communications, automatic rendezvous and docking, power switching, and crew avionics. For communications, three topics must be discussed. The size of the antenna(s) needed for adequate information exchange must be found. The power requirements for the system is to be shown. Also, the way information (telemetry) is relayed must be discussed. For rendezvous and docking, the relationship of automated technology to space station control is analyzed. The sphere of influence that the station has needs to found in regards to the role that expert systems play in docking. Moreover, the data requirements needed should be shown. For power switching, commands from mission control needs to be transmitted to the various subsystems. Power requests from the other subsystems should be received, evaluated and met. For crew avionics, the level of crew interaction must be determined.

Method of Attack

In the area of communications, the main design goal is to find the sizing of the antenna. While complete details can be found in the Appendix, a non-technical outline will be given now. First of all, the maximum data rate required for the system must be found. From research, it was discovered that the information flow needed for rendezvous and docking fixed the upper limit. Using the antenna equations, the size of the antenna(s) used are quickly found for assumed operational conditions. Secondly, the format of the telemetry must be chosen from various methods found in research. Finally, existing systems must be analyzed to see if the meet the mission requirements.

The first consideration under rendezvous and docking is to find existing systems and also related technology. Then, the protocols associated with close proximity operations around the space station

were found. Application of artificial intelligence was then looked at in light of positive space station control.

For power switching, existing systems were first researched to see if they met requirements. After this, power requirements for the chosen system were found.

For crew avionics, the level of desired crew interaction was first assessed. From that discussion, human interfaces for that system were then analyzed.

Design

Communications

Upon much evaluation, a modified version of the space shuttle communication system was chosen. Details of the space shuttle system can be found in Refs. 1 and 2. The system was based on the shuttle's for several reasons. Although it is a little dated, it is a proven, existing technology. Spare parts are obtainable from shuttle system's inventory.

The system consists of five antennas with the related processing equipment. Four of them operate on the S band (1.55-5.2 GHz). They are flush mounted and spaced 90° apart. By covering them with a tough dielectric, they can still receive and transmit while being partially protected from re-entry. The fifth operates on the K_u band (12-18 GHz) and is placed on a movable platform with associated position sensors. Tracking is determined by the sensors and controlled by actuators (Refer to Attitude and Articulation Control, AACS). Both types of antennas are compatible with space station communications. Also, they can interact with the TDRS (Tracking and Data Relay Satellite) system (see Fig. 1). It was decided to use the signal protocols already established for the shuttle in dealing with both the station and TDRS. Weights and volumes were found and relayed to Mission Management (MMPC), Power and Propulsion (PPS), and AACS so that the necessary calculations could be performed. Also, power requirements were given to PPC.

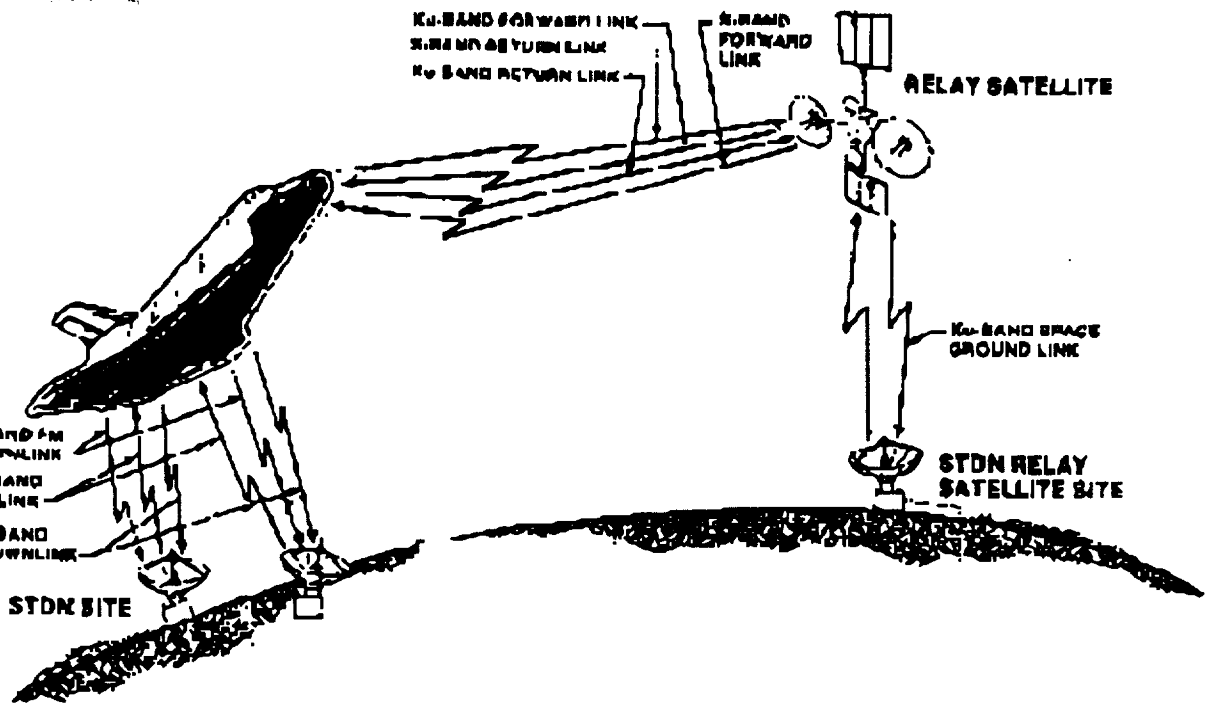

Fig. 1 Communication network

from Proceedings of IEEE, Vol. 75

Automatic Rendezvous and Docking

Two systems were the primary choices for this task: microwave interferometry or a shuttle derivative. The interferometer (Ref. 3) system uses a pulse radar to measure the relative angles between two spacecraft. The interferometer makes accurate assessments of the angular location of the spacecraft. Diagrams and a brief description can be found in the Appendix.

However, the system chosen was based on what the space shuttle employs (Ref. 1). It uses the same antenna that is used for K_u band communication with station and TDRS. Although it has not been actually employed on a space station the research has been, or is being, conducted.

From research (Ref. 2), it was found that the space station commands a thirty seven kilometer radius zone. In this zone, the spacecraft must have requests for attitude movements confirmed by the station. Artificial intelligence controls on the craft must then

work closely with the computers in the station in order to rendezvous and dock properly.

Power Switching

Since two of the subsystems already have roots in the space shuttle, it was fitting that the computer system that deals with them matches. While full details of the shuttle system can be found in Ref. 4, a brief description will be given. Each unit handles all the different subsystems of the craft. They work in parallel and are useful for redundancy. But unlike the shuttle, the system comprises only three units (see Fig. 2).

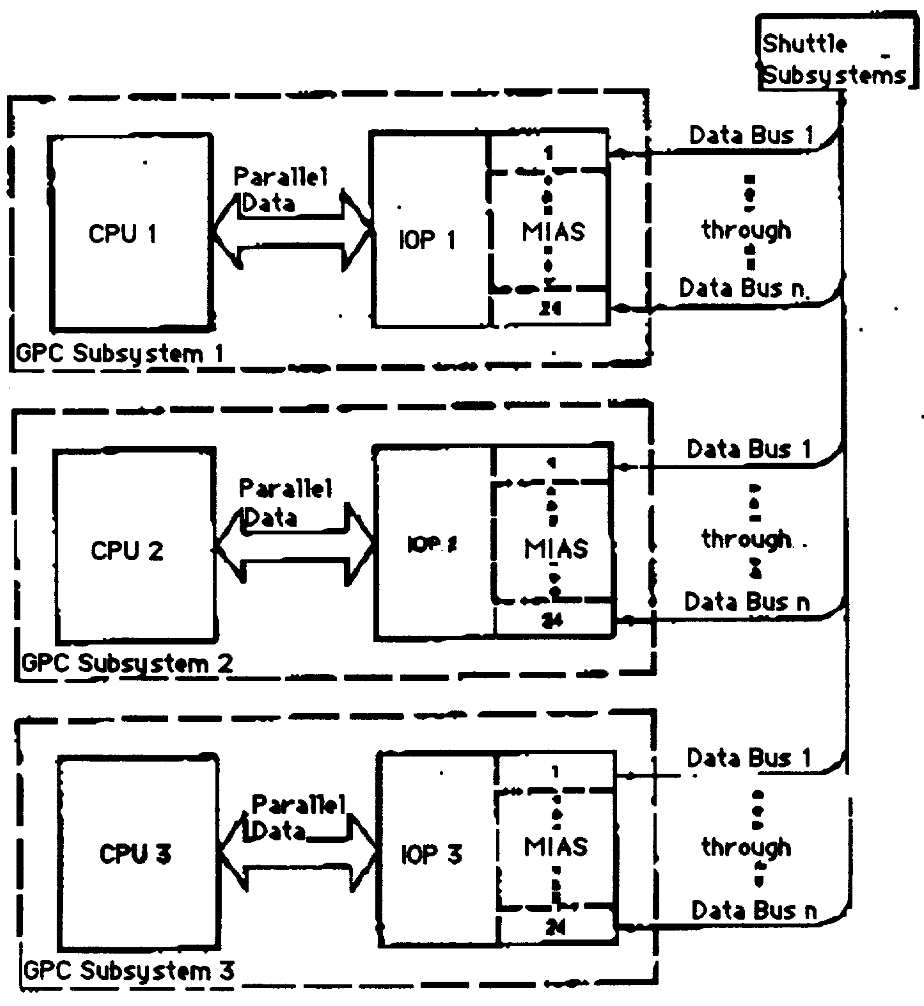

Fig. 2 Data Processing System

from Proceedings of IEEE, Vol. 75

Crew Avionics

Unlike the other subsystems, the role of human interaction appears to be an arbitrary decision. Crew avionics would only be present for psychological assurance of the crew. The routine operation of this craft is calls for automatic maneuvers with no crew on board. Only in emergency situation, such as injured crew or evacuation, would people be present. But then the question arises about injured crew trying to pilot the craft. If that situation occurs the craft should still be able to be controlled by the station or ground control. Therefore, for these reasons, it was decided that crew avionics would be not emplaced.

Concluding Remarks

Although the systems described above fulfill the requirements, it should be noted that their technology level is mostly dated. The communication and computer systems are 1970 technology. Further improvements have reduced the weight and power requirements for these systems.

Another area that still needs development is AI, artificial intelligence. Although progress has been made on expert systems and neural networks, the systems to deal with docking are still in development. When the shuttle first docks with the station, on the field experience will be gained.

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Appendix

Antenna Sizing

From research (Ref. 5), Shannon's Law relates information capacity with the power received.

$$B = W \log_2 \left(\frac{P_R}{P_N} + 1\right) = \text{info cap. (bits/sec)}$$

where: W = bandwidth (Hz)
P_R = received power
P_N = noise power

$$P_N = \frac{c}{kT} = \frac{1}{kTW}$$

where c = speed of light
k = Boltzmann's constant
T = Temp (°K)

Further equations relate received signal power, transmitted power, and antenna size.

For parabolic dishes: $P_R = P_T \left(\frac{4\ c\ D}{f z d_r d_t \pi}\right)$

For isotropic dishes: $P_R = \left(\frac{z\ A}{4\ \pi\ D^2}\right) P_T$

where: P_T = transmitted power
f = frequency
z = efficiency
d_r = diameter receiver
d_t = diameter transmitter

Using these equations and known constant values, shuttle designers made their system. The antennas have a diameter of .91 meters and can carry a data rate as high as 100 Mbits/s with sophisticated modulation.

Interferometer

From Ref. 3, "The position of the target vehicle is determined by measuring line of sight range and angles to the target vehicle (see Fig. 3). The relative attitude of the spacecraft is determined by measuring line of sight range and angles to four passive target aids symmetrically displaced about the spacecraft docking port (see Fig. 4). The target aids consist of passive, broadbeam antennas terminated in delay lines each having a different time delay. The relative pitch, yaw, and roll of the target vehicle is then calculated with the help of the guidance computer...A phase interferometer (see Fig. 5) is employed to measure line of sight angles to an accuracy of a few thousandth of a degree...The system described does not require any scanning of the radar antenna and provides instant acquisition of the four target aids." (Figures from Ref. 3)

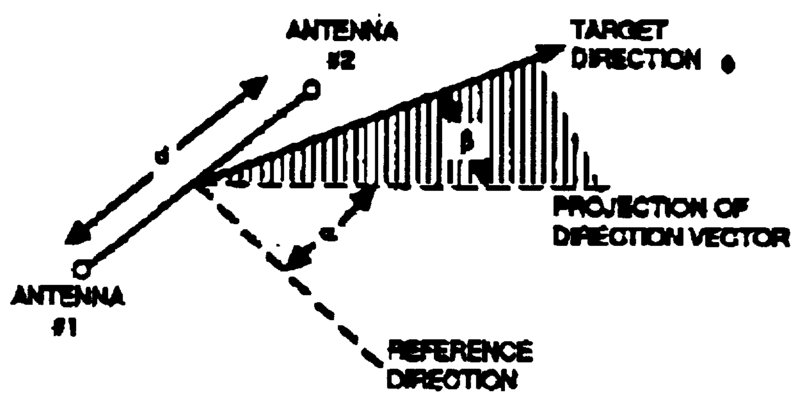

FIG.3 INTERFEROMETER GEOMETRY

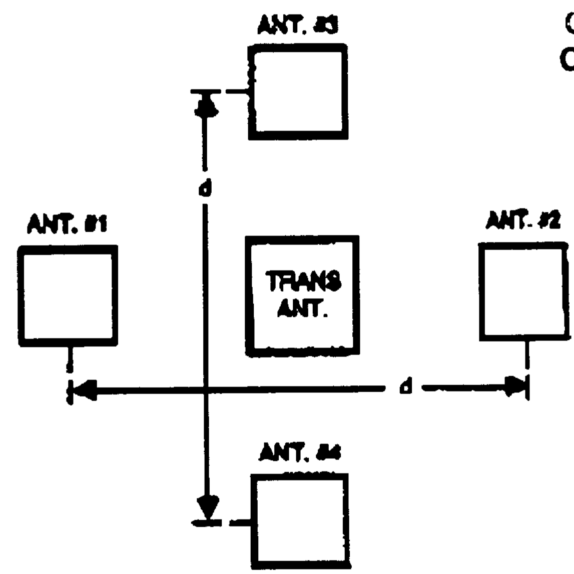

FIG. 4 DUAL INTERFEROMETER ARRANGEMENT

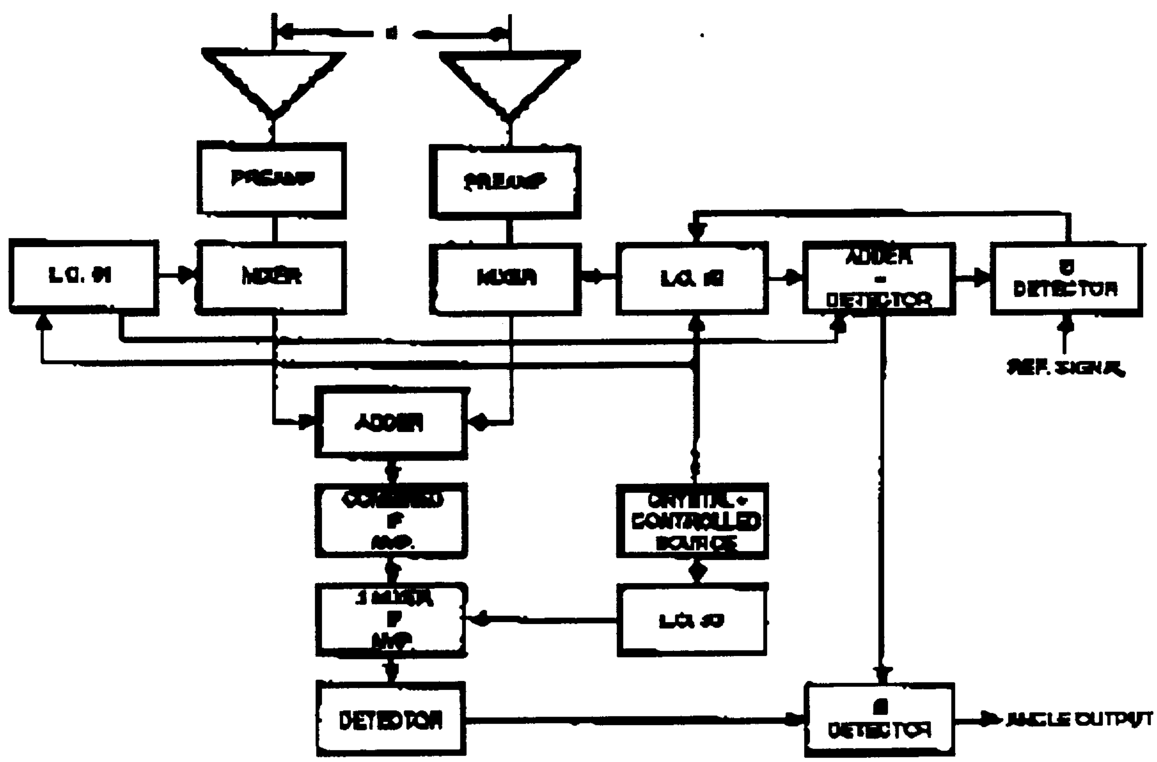

FIG. 5 BASIC INTERFEROMETER BLOCK DIAGRAM

REENTRY/RECOVERY SYSTEM

JOHN SELMARTEN

Introduction

The Reentry/Recovery System governs the vehicle from the time of departure from the Space Station until it has safely landed on the Earth and has beeen recovered. The basic driving requirements for the RRS are protecting the payload (or crew in emergencies) from both the gravitational forces and the thermal loads during atmospheric reentry. These requirements form the basis for the structural shape of the vehicle and the thermal protection system utilized. The RRS must provide a suitable reentry trajectory, suppplying the Propulsion System with a delta V needed for deorbiting from the Space Station orbit. The RRS must also provide a minimum of two landing sites, land or water, that are within one hour of medical facilities in the event of a medical emergency. The time of reentry must also be kept to a minimum in the event of crew medical emergencies, with an upper limit of 24 hours. Considering these driving requirements, along with others identified later, ARC meets all of the requirements with an efficient initial design.

Deorbiting

Space Station Freedom will orbit the Earth at a 28.5 degree inclination, the same inclination as Kennedy Space Center where most launches will occur. The altitude of the Station varies between 290 and 430 km, depending on solar cycles and position in orbit. Once waste products or crew are ready to return to Earth, the vehicle will undock from the Space Station via the cold gas thrusters described in the AACS. The vehicle will move to a stationary position, relative to the Space Station, at a distance of about .5 km. This is needed so any engine firing does not affect the Space Station's immeadiate environment. This co-orbiting condition will remain until the reentry window is available. Since the orbital period of the Space Station is 1.553 hours maximum (calc. R-1), this procedure will take about .2 hours for routinely planned descents and a maximum of .8 hours in an emergency given two opposing reentry windows.

The trajectory described is very simple due to the fact that the resources needed to optimize the dynamic flight

path requires computer coding and atmospheric modeling too complex to be performed at this time. However, some assumptions can be made to give a fairly accurate model. Using a very simple Hohmann transfer from the Space Station to the Earth, the delta V needed is .1258 km/s (calc. R-2). For safety reasons, .14 km/s of fuel will be allowed in the event of a last minute change in trajectory. The velocity of the vehicle when it first encounters the traditional boundary of the atmosphere (122 km alt.) will be 7.886 km/s (calc. R-3) and the angle of attack coming into the atmosphere will be less than 2.4 degrees (calc. R-4). To keep the weight of the fuel to a minimum, this trajectory will occur with no plane shift maneuvers, since they are very costly in fuel requirements as shown by equation R-5. Therefore, the landing crossranges are important and must allow for different landing sites.

Atmospheric Reentry

The period when the vehicle is between 120 and 18 km alt. is the most important. During this time the vehicle must slow from hypersonic flight to subsonic. The friction between the vehicle and the surrounding air molecules slows the vehicle's vertical speed to that of its terminal velocity, the fastest that it can pass through the increasingly dense atmosphere. This speed of less than Mach 1 is usually attained by 18 km alt. (50,000 ft.) and is due to the size and shape of the vehicle. Generally, communication during this portion of the flight is impossible. The great amount of heat generated ionizes the surrounding air, creating enough electrical interference to block any form of communication. The vehicle must be preprogrammed to perform the correct attitude adjustments to compensate for the heat. This is accomplished through interaction with the Command and Data Control and Attitude and Articulation Systems.

Several driving factors during this phase of the mission contribute to the shape of the vehicle. The g forces experienced by the payload must be minimized, and the crossrange capabilities must be large enough to permit an adequate variety of landing sites. The vehicle will be in a 28.5 degree inclination orbit. Figure R-1 shows a ground tracking of the vehicle. For the preferred land based landing site, a significant crossrange capability must be designed into the vehicle. Blunt bodies do not provide the required crossrange needed; they drop ballistically. A lifting body design can provide much more acceptable crossrange capabilities. Figures R-2 and R-3 show the crossrange and g forces as functions of the lift to drag coefficient of the body. From the requirement to minimize

the g force and maximize the crossranges, a L/D coefficient of about one seems optimal. While numerous vehicle configurations have been researched, our design is based on the requirement to have an L/D ratio of about one during reentry to keep the g forces less than two. This design is illustrated in figure R-4.

Thermal Protection

During the atmospheric reentry, thermal loads are experienced due to the friction between the supersonic vehicle and the air molecules. Figure R-5 shows how the density of the atmosphere increases rapidly during descent from the 120 km boundary. The thermal loading is a dynamic function of the vehicle shape, angle of attack, velocity, and density of the atmosphere. The lifting body design allows for a control of the angle of attack via the rear flap. This gives some control over the heat load, but thermal protection must be included. This thermal protection must capture and dissipate enough of the heat produced to keep the vehicle's aluminum structure to a temperature less than 200 degrees Celsius. This requirement is due to the structural limits of the aluminum as given by the Structures System. For the design suggested here, a maximum heating rate of 200 BTU/(s ft^2) was set.

Figure R-6 shows several different materials which can be used for a protective coating. Since a major requirement of the design is reusability, ablative thermal protection can not be used efficiently. Ablative materials would require reapplication after each mission, creating very high costs and turnover time. A reusable thermal protector is preferred. It is important to keep the weight low due to the propulsion considerations, therefore, a low weight/surface area is desired. An optimum material choice is LI 900. This is the same ceramic material used on the space shuttles and is therefore a proven technology.

Because of the geometry of the craft, the heat loads are concentrated at a few critical areas. The nose cone must be made with a greater thickness than the cylinder because of the more concentrated heating. The tip of the nose and the moveable flap must be made of a much more heat resistant material than LI 900. For these areas, a carbon-carbon compound has been chosen because of its excellent thermal properties.

Landing

Several different methods are available to slow the vehicle from its terminal velocity to the less than 5 ft/s needed for a safe touchdown. Aerobraking is one form of deceleration, however, several drawbacks surface when considering it. A reasonable crossrange is needed so extra fuel is not needed for orbital plane changes, but the aerobrake method causes the vehicle to drop ballistically. It does not work easily with the lifting body design. Also the technology associated with aerobraking is relatively new, and it has not yet been field tested; the technology may not be available by the 1994 date set in the RFP. Retro-rockets, such as those used by the lunar landing crafts, are not considered because of the large weight and bulky system design necessary to produce effective deceleration. Parachutes have been used for almost all Earth reentry vehicles which have been designed for recovery, therefore, they will be considered.

Simple conical parachutes have been used exclusively by the government to date. They provide the needed deceleration for water landings, however, they are uncontrollable. The point of touchdown is completely dependent on prevailing winds. Also the final vertical velocity is greater than the 5 ft/s needed to protect the equipment or crew from shock for a land landing.

A better parachute system is the rectangular parachute. This type of parachute allows the payload to be very maneuverable, thus further increasing the crossrange potential. Figure R-7 shows a sample crossrange enhancement that can be achieved thru the use of a rectangular parachute. Because of the great amount of maneuverability, prevailing winds can be compensated for and accurate landings can be realistically achieved. Another advantage is flareout. This is the dynamic action of applying a large, impulsive force to both rear suspension lines causing braking, which provides a temporary zero vertical velocity. If this procedure is used at the proper time, right before touchdown, the shock of landing can be reduced significantly.

For this type of parachute a mechanical control mechanism must be employed to work the suspension lines, providing the maneuverability. This mechanism requires about 5 Watts of power and can be remotely controlled from the ground via a visual, handheld control unit or an autonomous computer control signal. Since the computer control does not rely on visual confirmation or human judgement, and can benefit from a vast amount of atmospheric conditions updated every few seconds, it will be the form of control utilized. While this mechanism can be weighty (about 500 lbs. for mechanics, signal receiver, and power source), the rectangular parachute itself

requires less surface area than a comparable conical parachute. The reason for this is rectangular parachutes are aerodynamically designed to perform optimally for a given requirement. The conical parachute is simply designed to slow vertical descent, while the rectangular parachute performs similar to a glider, giving the operator horizontal as well as vertical control through actuations of the suspension lines. Conical and rectangular parachute systems have very similar weights for payloads of about 5,000 lbs., but as the payload increases, the rectangular system weighs less due to the fact that the control box does not increase in size or weight for larger sized parachutes: it is a fixed unit. Also many of the instruments required for the control box are already available on the vehicle. For example, the power source and signal receiver have already been designed by the Power System and the Command and Data Control System and can be used instead of an independent electronics package specifically for the parachute system.

The technology for use of a remotely controlled, rectangular parachute system is relatively new. Several tests involving payloads of 600 and 1500 lbs. have been successfully completed (Ref. 2), but for use on the ARC the payload limit must be about 70,000 lbs.. This doesn't seem to present a great engineering problem, since it only involves scaling up present canopy configurations and increasing the number of suspension lines. The only real problem would arise from the need to adequately field test the system on payload of similar weight and shape of the vehicle. Testing of systems of this weight can be difficult due to the need to drop the test from an altitude of at least 50,000 ft., though this expenditure would provide a useable system that would more than pay for itself in the long run.

Figure R-8 shows a conceptual depiction of the parachute to be used on ARC. Once the vehicle has slowed to less than Mach 1, at about 60,000 ft alt., a pilot chute will be released. This pilot chute will extract a drouge chute placed in the rear of the vehicle. Its purpose is to slow the vehicle both horizontally as well as vertically. When the vehicle has reached an altitude of about 50,000 ft., the main canopy will be released from an area on the top and very close to the center of mass of the vehicle. Shortly thereafter, the drouge chute will be released. In the event of failure of deployment or complete loss of parachute, the drouge chute will not be released, but will be used along with the cold gas thrusters to slow the vehicle enough for an emergency water landing. This is only to be used as a last chance option. The vehicle will first attempt several roll and pitch maneuvers to inflate the canopy if it is not fully inflated. It is possible to land with only 60% of the suspension lines intact, but

maneuverability is drastically cut and flareout is generally not possible. The idea of rectangular parachutes is not a new technology, skydivers have used them for over a decade. The only new technology involved is scaling up the concept for a much larger payload and incorporating a mechanical control mechanism. For use on ARC, a rectangular, remotely guided parachute system is ideal when considering both the weight and controllability.

Landing Sites

The major considerations when selecting landing sites are the requirements to be within one hour of emergency medical treatment and to keep costs and turn around time to a minimum. These requirements indicate that a land based site is preferable. A water landing would necessitate a costly naval recovery fleet for every routine landing. Also, the corrosive properties of salt water make routine landings very costly in terms of either protective paints or replacing damaged parts. A land based landing allows for minimal cost when the vehicle must be returned to the warehouse for structural and electronical tests after every mission. Different modes of transportation from the landing site to the warehouse would include train or cargo airplane depending on the availability of either.

Because of the trajectory, the only available continental U.S. landing sites are in the South. Initially, the wide open space, such as in the Southwest will be used until the maneuvering parachute has performed with enough reliability to permit landing in the more congested areas nearer to the launch pad, Kennedy Space Center in Flordia. More than one landing site must be available because, in the event of a complete Space Station evacuation, two vehicles will be reentering the Earth at basically the same time. Also in the event of some emergency at one site, such as fire or severe storm, other sites can be utilized. Military bases will be augmented, such as Edwards Air Force Base and White Sands, in order to reduce the cost of building new landing facilities. Each of these bases has its own medical facilities and existing personnel which could be prepared for the routine landings every three months or the rare, unscheduled emergency landing. An additional emergency landing site could be prepared in Australia to reduce the time to touchdown in the event of an emergency occuring after the reentry window for a U.S. landing has passed. This site would not be used for routine logistics landings because of the high cost of transporting the vehicle and its cargo back to the repair warehouse or launch pad.

Since the rectangular parachute is controlled by a homing beacon, it will land very close to its target. The point

of contact with Earth must be prepared by clearing any obstacles in a 5,000 square foot area to prevent any collisions. By preparing the ground to absorb some of the shock, the vehicle does not need to be outfitted with any weighty shock absorbing gear. The ground should be filled with loose dirt or some other form of shock absorbing material, depending on how effective the flareout is at reducing the touchdown velocity. A suspension system shown in figure R-9 could be used to capture the parachute and lower the payload using viscous-damping tension cables. This would allow ARC to land with little or no shock damage to the structure, equipment, or crew and at the same time prevent the canopy from tangling or covering the escape hatch so medical treatment can be administered as soon as possible.

Summary

The basic reentry scenario is to detach from the Space Station via cold gas thrusters, and remain orbiting the Earth with the Space Station at a reasonable distance. During this time the vehicle will attain the proper attitude for reentry engine firing. When a reentry window is available, as determined by mission control on the ground, the main engine will fire the required amount of delta V to slow the vehicle and place it in a reentry orbit. Once the engine has stopped firing the vehicle will again adjust its attitude so it is properly positioned for reentry. The vehicle will continue in this orbit until the atmosphere begins to change dramatically at 120 km altitude. Once in the denser atmosphere, the lifting body design will be utilized to reduce the heating and g forces, and deliver the needed crossrange for the intended landing site. The information for these adjustments must be preprogrammed into the vehicle's computer since ground communication during this phase is impossible. Once the vehicle has reached an altitude of 18 km, its vertical descent speed will have decreased to subsonic and the pilot chute will release the drouge chute. From this point until touchdown, the vehicle will be under the control of the landing site computers. At about 15 km, the main rectangular parachute will be deployed and if there are no problems, the drouge will be detached. The parachute will be guided down to the landing site by the ground computer which has the necessary atmospheric data (including wind velocity and temperature) updated at a reasonable rate. Just before touchdown, flareout will be employed to reduce the vertical speed to near zero. At the moment of flareout, the canopy will be captured by the suspension structure based on the ground and the vehicle will be slowly lowered to the shock absorbing material prepared on the ground. For routine logistics landings, the cargo will be

removed and shipped to its respective laboratories within a few hours. For emergency crew return, the crew will be removed immeadiately by trained medical personnel who will determine whether they can be transported to nearby medical facilities. In either case the vehicle will be shipped back to a repair warehouse within a few days, where it will be examined and outfitted for the next mission.

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R-10 Tewell, J.R. An Unmanned Reentry/Recovery Vehicle for a Reusable Launch System, AIAA Paper 84-0781, AIAA 8th Aerodynamic Decelerator and Balloon Technology Conference, Hyannis, Mass. April 2-4, 1984.

CALCULATIONS

Constants:

$u = 3.986 \times 10^5 \ km^3/s^2$

$R_{(Earth)} = 6378$ km

Calc. R-1: Orbital period

S.S. orbital period at 290 km alt. = $T_{(290)} = (2pi)*(a^3/u)^{1/2}$

assume a circular orbit so a = r = (R + alt.)

$T_{(290)} = (2*pi)*((6378+290)^3/3.986 \times 10^5)^{1/2} = 5419$ s

$T_{(290)} = 1.5$ hours

$T_{(430)} = 5590$ s = 1.553 hours

Calc. R-2: Circular to Hohmann delta V (worst case - 430 km alt.)

$V^2_{(circ)} = u/r = u/(6378+430)$

$V_{(circ)} = 7.6517$ km/s

$V^2_{(Hoh)} = u*(2/r - 1/a)$

where a = (6378 + (6378 + 430))/2 = 6593 km

$V^2_{(Hoh)} = u*(2/(6378+430) - 1/6593)$

$V_{(Hoh)} = 7.5292$ km/s

delta V = $V_{(Hoh)} - V_{(circ)} = -.1258$ km/s

Calc R-3: Velocity during Hohmann transfer - atmospheric entry

$r_{(122km\ alt.)} = 6378 + 122 = 6500$ km

$V^2_{(entry)} = u(2/6500 - 1/6593)$

$V_{(entry)} = 7.886$ km/s

Calc R-4: Angle of attack (@) - atmospheric entry

$\cos @_E = (a^2*(1-e^2)/r*(2*a - r))^{1/2}$

where e = (r(apogee)/a) - 1 = ((6378 + 430)/6593) - 1

e = .03261

$\cos @_E = ((6593^2)*(1 - .03261^2)/6500*(2*6593 - 6500))^{1/2}$

$\cos @_E = (.999135)^{1/2} = .9995676$

$@_E = 1.685$ degrees

Calc R-5: Orbital plane changes

delta V = 2*V*sin(delta O/2)

O = the angle of inclination

For small delta O, the delta V is approximately twice the velocity multiplied by the change in orbit inclination. As the delta O is increased, the delta V becomes unreasonably high.

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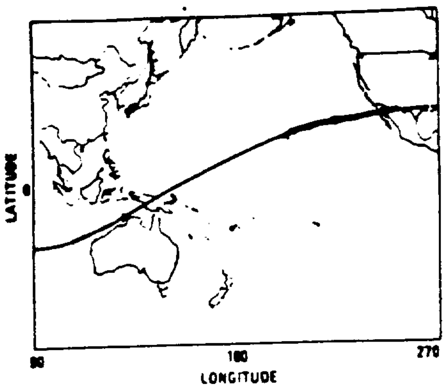
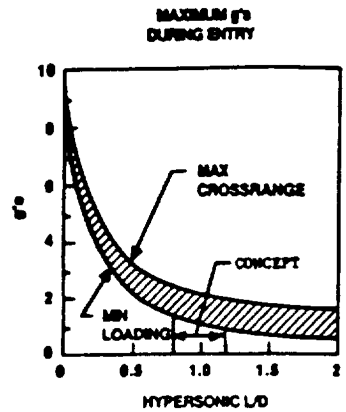
FIGURES R-1 – R-9
REENTRY/RECOVERY SUBSYSTEM



Recovery Events for a KSC Launch

R-1
ref R-10



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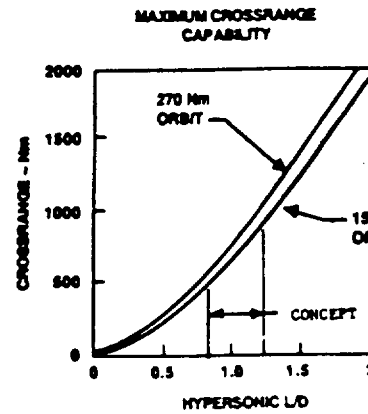

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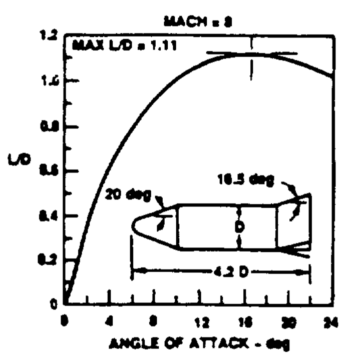

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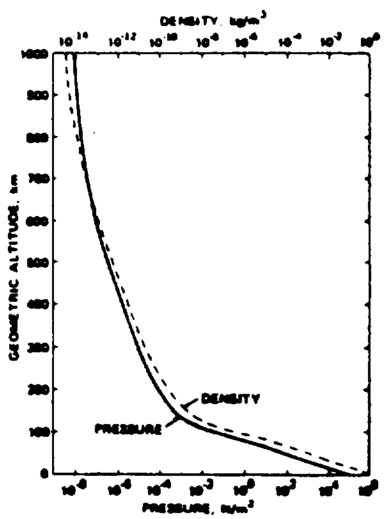
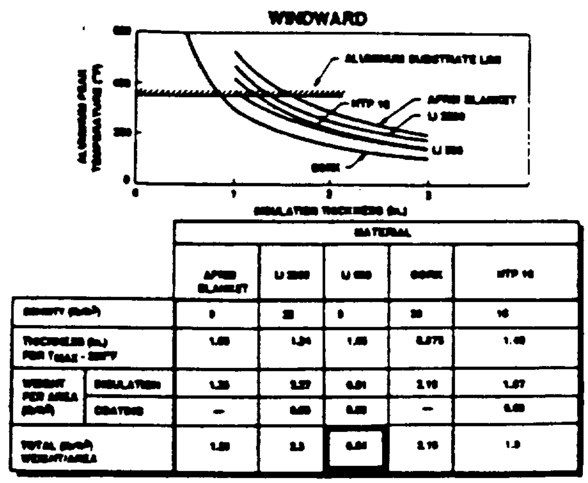


R-5
ref R-7



R-6
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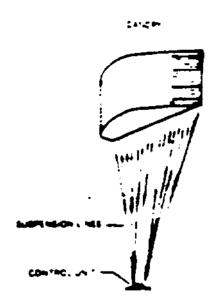
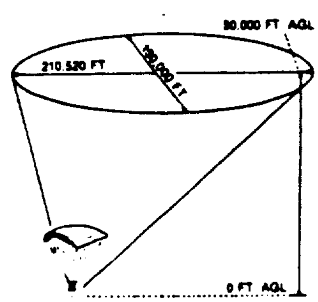


RAM-AIR INFLATED DUAL MEMBRANE GLIDING PARACHUTE

- PAYLOAD WEIGHT 70,000 LBS
- TOTAL RECOVERY SYSTEM WEIGHT 3,135 LBS
- PLANFORM RECTANGULAR
- CANOPY AREA 12,950 FT²
- ASPECT RATIO 3:1
- SPAN 194.25 FT
- CHORD 66.67 FT
- CELL WIDTH 22 IN
- AIRFOIL SECTION LISSAMAN 7808
- WING LOADING 5.5 LBS/FT²
- BASELINE LENGTH 105 FT
- CANOPY LINE ATTACH POINTS 3,584 CASCADED TO 112 LINES AT THE RISERS
- RISER LINE ATTACHMENT POINTS 112 CASCADED TO 4 POINTS ON PAYLOAD
- PACK VOLUME 40 FT³

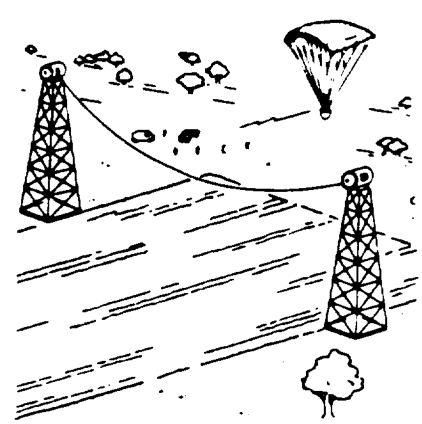
R-8
ref R-5



TYPICAL CONTROLLED RANGE ENVELOPE

REPRESENTATIVE WIND PROFILE
(CANADA JUNE 13, 1984)
30 KTS AT 50,000 FT
45 KTS AT 25,000 FT
10 KTS AT 1,000 FT
CALCULATED (0-50,000 FT) AT 1,800 FT INTERVALS

R-7
ref R-5



LAND BASED ARRESTING GEAR RECOVERY

R-9
ref R-5

STRUCTURE SUBSYSTEM

Steve Hermann & Mark Mueller

The primary function of the ARC structure is to provide mechanical support to all the subsystems within the framework of the spacecraft configuration. The structure must also satisfy various requirements, such as docking capability, component layout, payload size, thermal control, reentry aerodynamics, and launch vehicle compatibility. The spacecraft will be subject to major mechanical loads during launch and must be designed to survive the launch loads and to protect the other subsystems. While in space, the loads will be significantly lower than the launch loads, however, the structure must possess a high stiffness for the deployed appendages to avoid interaction with the attitude control system.(Agrawal, 1986 p.179) These requirements are the driving factors in determining the size, shape and weight of the ARC and also help specify what materials will be used. Throughout the development and designing of the ARC, the structure fabrication ease and safety factors must be taken into account.

STRUCTURE SHAPE AND SIZE

The shape of the ARC was chosen by the Reentry and Recovery Subsystem due to the aerodynamic features necessary for reentry into the atmosphere. The necessary volume of the ARC was estimated using the estimated volumes required for each of the other subsystems. Then, with the dimensions of the chosen launch vehicle, the size of the ARC was calculated. The overall dimensions of the ARC may be seen in Diagram 1 of the appendix of this section. The ARC structure has a cargo capacity of 70.0 m^3 and can carry 16500 kg of payload mass. Due to the weight restriction imposed by the launch vehicle, however, the ARC can only take up a maximum of approximately 11500 kg. The actual liftoff weight is listed in the Mission Management, Planning and Costing Subsystem.

STRUCTURAL MATERIAL

The body of the spacecraft will be constructed of various beryllium aluminum stringers and bulkheads. The actual panels will be made of an aluminum honeycomb core and beryllium lockalloy skins.(Agrawal, 1986, p.242) This dual wall system will protect the crew and subsystems components from possible micrometeorite impact. This layout is shown in Diagram 1 of the appendix. The beryllium lockalloy was chosen as it combines the ductile properties of aluminum with the higher strength properties of beryllium. It has a high modulus, low density, high formability, and good machining characteristics. This alloy, developed specifically for space structures, also

exhibits useful structural properties in the 315 to 425 ^{o}C service temperature range.(Agrawal, 1986 p. 249). This provides the necessary thermal control for the system.

To protect the structure form the higher temperatures of reentry, ceramic tiles like those used on the shuttle, will be employed. The front cone and the back flange will require a 3.5 inch thick tile to sufficiently protect the structure. The main, cylindrical body requires only one inch thick tiles. On the extreme nose of the ARC, a carbo phenolic heat shield will be used. The flap at the end of the ARC will be constructed of a carbon-carbon material. These materials were prescribed by the Reentry and Recovery Subsystem. The layout of these structural materials can be seen in Diagram 1 of the appendix.

STRUCTURAL APPENDAGES

Several components are required to be attached to the structure. To fulfill the requirement of docking capability, a docking adaptor must be employed. Specifically, the docking adaptor must be the one used by the space station. The details are shown in Diagram 2 in the appendix.

The Command and Data Control Subsystem requires four, one meter antennae to be located on the front con of the ARC. Their location can be seen in the Design Layout section. Another requirement is that of a movable parabolic antenna. The placement of this antenna is described within the Attitude and Articulation Control Subsystem.

The Power and Propulsion Subsystem and the Attitude and Articulation Control Subsystem require an extendable solar array. Systems of this type have been developed for other space missions. Diagram 3 in the appendix shows the operation of such a system. The stowed volume was approximated as a 0.50 m in diameter cylinder stretching across the diameter of the ARC. The weight of the extendable arm was approximated by a 0.3(m in diameter cylinder of 2 cm thickness beryllium lockalloy. The required length of the arm when fully extended is 9 m.

SUMMARY

The ARC structural shape was chosen within the Reentry and Recovery Subsystem. The main structural materials are aluminum and beryllium lockalloy. Ceramic tiles, carbon phenolic material and carbon-carbon are employed as prescribed by the Reentry and Recovery Subsystem for thermal control. The ARC is designed to carry a cargo of 16500 kg in a volume of 70.0 m^3. To remain compatable with the chosen launch vehicle,

however, only approximately 9000 kg of payload may be taken up. The component layout and calculation of the center of mass of the design is shown in the Design Layout section.

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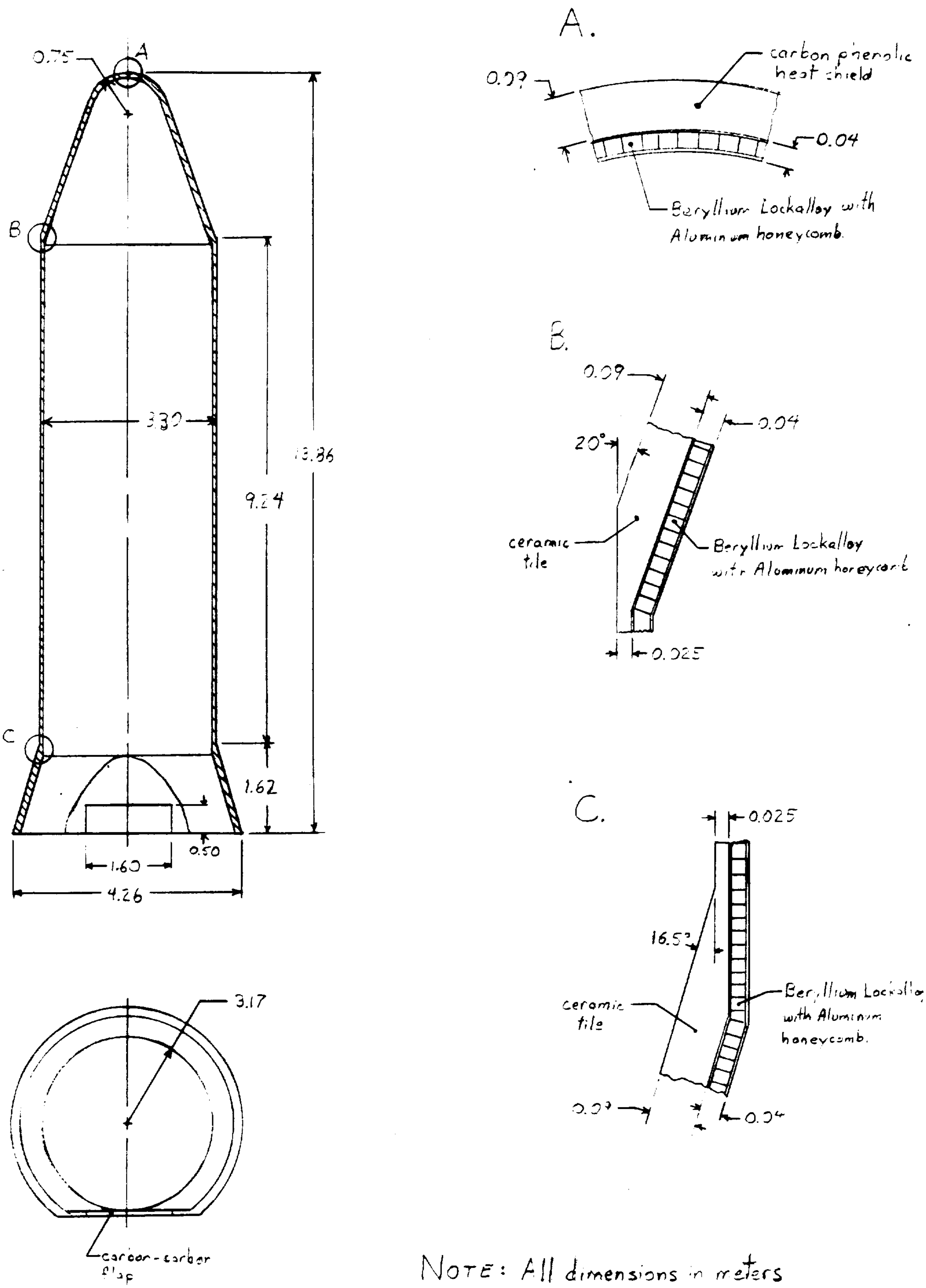
DIAGRAM 1

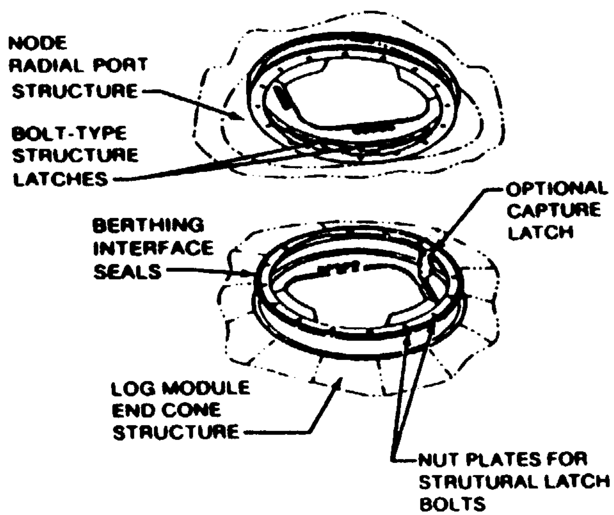
Structural Ske eton and Dimensions

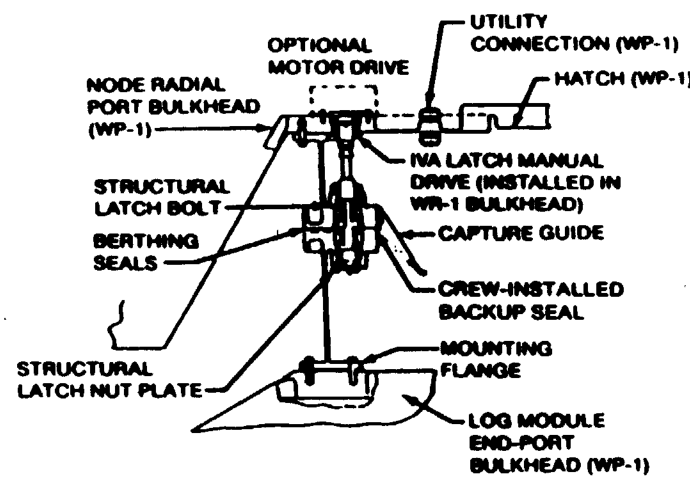


NOTE: All dimensions in meters

Structures Appendix

DIAGRAM 2*





PHYSICAL CHARACTERISTICS

NODE BERTHING HALF		SIZE (IN.)		
WEIGHT (LBS)	371	• HEIGHT		7.25
• STRUCTURAL FRAME	203	• LATCH INSTALLATION DIA		70.7
• LATCH BOLTS AND MANUAL DRIVE	90	• BERTHING	WITH GUIDE)	60.5
• MISC COMPONENTS	78	RING ID	(WITHOUT GUIDE)	66.5
		PASSIVE (NO POWERED COMPONENTS)		

LOG MODULE BERTHING HALF		SIZE (IN.)	
WEIGHT (LBS)	344	• HEIGHT	10.8
• STRUCTURAL FRAME	218	• CYLINDER ID	74.0
• LATCH NUT PLATES	48	• BERTHING RING OD	77.0
• MISC COMPONENTS	78	• MOUNTING FLANGE OD	78.2
		PASSIVE (NO POWERED COMPONENTS)	

* Taken from AAE 241 handout

DIAGRAM 3

LMSC SEPS ARRAY *

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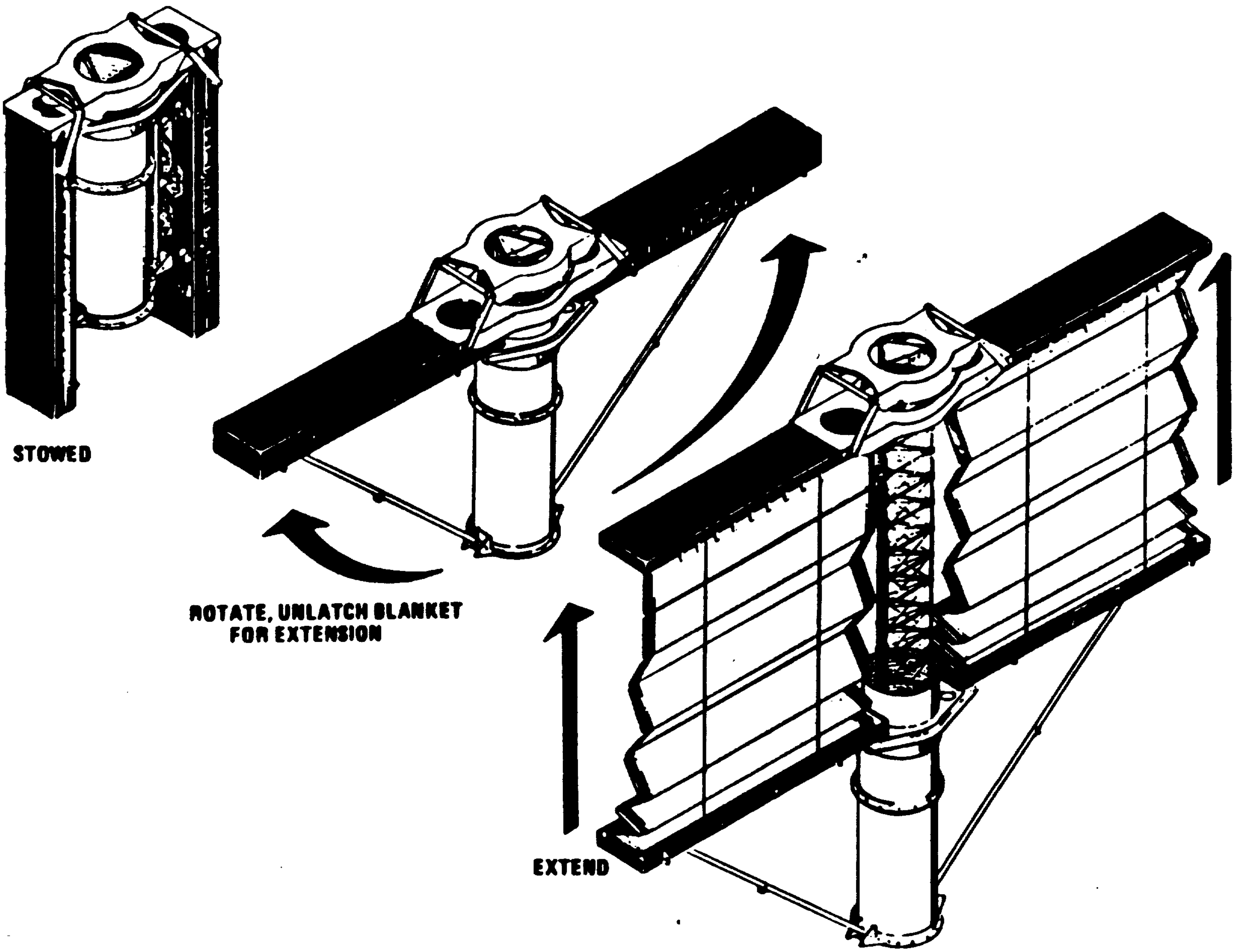

*AAE 241 Lecture Notes, pg 24

ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM

Mark Mueller

The Attitude and Articulation Control Subsystem is required to control the spacecraft attitude when travelling between the launch vehicle, the space station, the space platforms, and upon reentry. It is also required to perform the necessary docking maneuvers for attachment to the space station and to the space platforms. This subsystem also consists of actuation control for the sensors, communications devices, and the solar array. A system for payload loading and unloading is also described in this section. This section of the report is thus divided into the following components: attitude control, actuation control, and loading control.

ATTITUDE CONTROL

Attitude control may be attained by either a mass expulsion system, using reaction jets, or by a momentum exchange system, using either reaction wheels or control moment gyros (CMGs). Mass expulsion systems are well suited for maneuvering, but are mechanically complex, heavy, and are limited due to the fuel required. Momentum exchange systems do not use expendables, but rather power. These systems are also mechanically simple. Specifically, CMGs give high torque outputs, have low weight and low power requirements, and high pointing accuracy is possible.

Based on the above mentioned advantages and the requirement of a six year lifetime, control moment gyros were chosen as the main system for attitude control. The three configurations, shown in Diagram 1 of the appendix at the end of this section, were looked at. The three single-gimbal CMG system contains no redundancy in operation and contains inter-axis coupling of response. The three single-gimbal CMG pairs system minimizes the coupling, but is much heavier and bigger. The actual momentum utilization of each CMG is also only 33%. The Sixpac configuration is lower in weight than the paired system, has 100% redundancy, and utilizes 100% of the momentum of each CMG.(O'Connor, 1969, p228) For these reasons, the Sixpac configuration was chosen for the ARC attitude control system.

A double-gimbal CMG used in the Sixpac configuration is a two degree of freedom gyroscopic device which consists of a constant speed wheel held in an inner gimbal, which is coupled ot an outer gimbal through the pivot perpendicular to the wheel spin vector. The outer gimbal is held to the base by a pivot perpendicular to the inner pivot. Both pivots are driven by geared motor torquers.(O'Connor, 1969, p.228) The moment of the spin-ning gyros then creates the necessary torque required to adjust the space-craft attitude. A computer is then used for the

control law governing the CMG gimbal servos. It is proposed to use the computer selected within the Command and Data Control Subsystem for the execution of the control law.

Although the CMGs are effective for attitude control during flight, their use gives inadequate maneuverability for docking procedures. For this reason a reaction jet system was designed for the docking maneuvers. A secondary function of the reaction jets is that of CMG unloading. This improves the vehicle rate transient and overall efficiency of the system (Jacot, 1966, p.1317). The reaction jet system is small and thus does not bring up problems encountered with weight, mechanical parts, and fuel that a larger mass expulsion system would.

Since simplicity of design was thought necessary, a cold gas jet system was chosen. A layout of a typical cold gas system is given in Diagram 2 in the appendix. Typical exhaust velocities range from 500 to 1000 m/s with thrust values of .05 to 25N.(Hughes, 1968, 4-2) The lower efficiency of this system compared to others is outweighed by the simplicity of the design, since only a small system is necessary. The use of cold gas also allows for safe operation near the space station.

An approximation of the required fuel was calculated using gaseous nitrogen stored at 3000 psi and 80^o F. The I_{sp} was assumed to be 100 seconds. The required Δv was taken to be four times that required to break away from the space station as specified in homework #6, or .4876 m/s. The mass of the ARC was taken to be 22,500 kg, which is the maximum allowed for use in the chosen launch vehicle. Using equations 1 and 2 from Table 1 in the appendix, the mass and volume of fuel required may be calculated. According to W. G. Hughes in his book, Active Stabilization, the mass of the container for this gas may be twice as much as that of the contained gas. Using this as an approximation, the weight of the fuel and containers was estimated to be 33.6 kg.

Only two thrusters are proposed to be used. They shall be placed as shown in the Design Layout. Movement perpendicular to the jets may be accomplished by adjustment of the ARC position by the CMGs, firing of the jets, and then repositioning the ARC by the CMGs.

The position of the ARC must be ascertained in order for the proper signals to be sent to the attitude control devices. This is accomplished by two means. Star and sun sensors are used to find the initial position of the ARC. Since this data acquisition is slow, gyros are used for rate integrated information of the position. Rate integrated gyros, however, require updated information from primary sensors to correct for the drift offset inherent in the system.(Chobotov, 1989, p.9) The star and sun sensors are employed again for this purpose.

Due to the low accuracy of horizon sensors, it is proposed to use two star sensors and a sun sensor. A sun sensor is employed due to its simplicity and low weight and power requirements. The other two necessary primary sensors will then be stellar sensors. All three of these devices will be mounted on a retractable scan platform. This

configuration can be seen in the Design Layout section. For the rate integrated gyro system it is proposed to use Resonant-Fiber Optic Gyros (R-FOG). This is a newly developed device. It is felt that by 1994 this instrument will be thoroughly tested and perfected. R-FOGs are beneficial due to their extremely small size, approximately four inches in diameter, and low weight.(Klass, 1989, p.81)

ARTICULATION CONTROL

The Command and Data Control Subsystem requires a one meter parabolic antenna that must track the space station. Due to the requirement of an outside mounted, movable antenna, it was decided to use a retractable platform so that the ARC will retain the desired aerodynamics for reentry. The star sensors also need to track their target stars. It was decided to mount the antenna, star sensors and sun sensor on the retractable platform. The antenna and star sensors would then be individually pivoted by mechanical torquers to keep their desired orientation. Given the size of the required antenna the platform is designed to be one square meter in area. The location of the platform and devices is shown in the Design Layout.

The Power and Propulsion Subsystem requires the use of a solar array. Again, to retain the aerodynamics of the craft, the solar array must be retractable. This will be done mechanically since hydraulic systems are too large and heavy. The array arm will also rotate for best solar reception by use of mechanical torquers. The size of the array arm is further described in the Structures Subsystem.

LOADING CONTROL

The ARC is designed to deliver and return material to the space station and to the platforms. Loading and unloading at the space station could easily be done by the space station crew. A method of loading control must be developed, however, for rendezvous with the space platforms. Industrial robot technology is advanced sufficiently to allow the use of a robotic arm for the loading control. Conveyor belts would be impractical due to the low gravity. This leads to a choice between an arm of sufficient length to reach everywhere within the ARC or an arm on a track inside the ARC. A hydraulic system would lift heavier loads, but would be impractical within the ARC due to the large size and weight. A robotic arm mounted on the space platforms that could reach everywhere within the ARC would also seem impractical due to the length of arm required. It was thus concieved to use a mechanical robotic arm on a track within the ARC capable of moving any payload to the docking adaptor. The movements of the arm would then fall under the command of the computer chosen by the Command and Data Control Subsystem. Further loading control would then be the responsibility of the space platform. This design is purely conceptual.

No known vehicles have used such a system. A possible arrangement of this system is shown in the Design Layout.

SUMMARY

To fulfill the requirement of attitude control and docking maneuvers, the ARC uses a combination of a control moment gyro system and cold gas reaction jets. To satisfy the requirement for antenna and sensor pointing control, a retractable scan platform was designed on which the antenna and sensors are pivoted by mechanical torquers. Actuation control for the solar array consists of retracting the array arm and pivoting the array by mechanical means. Loading is performed by a track mounted, mechanical robotic arm.

References:

Chobotov, V.A., Lecture Notes: Spacecraft Attitude Dynamics and Control, AAE 391, 1989.

Hughes, W.G., Active Stabilization, Farnborough, UK , 1968.

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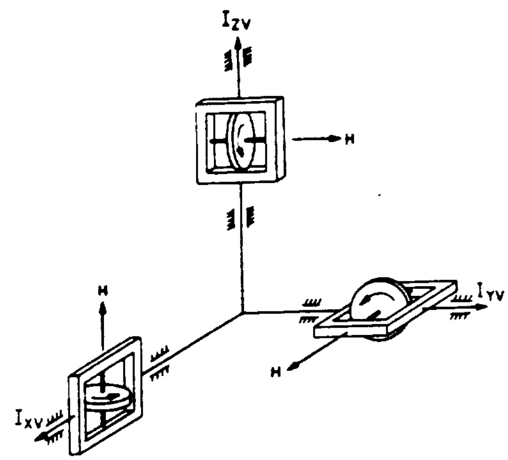
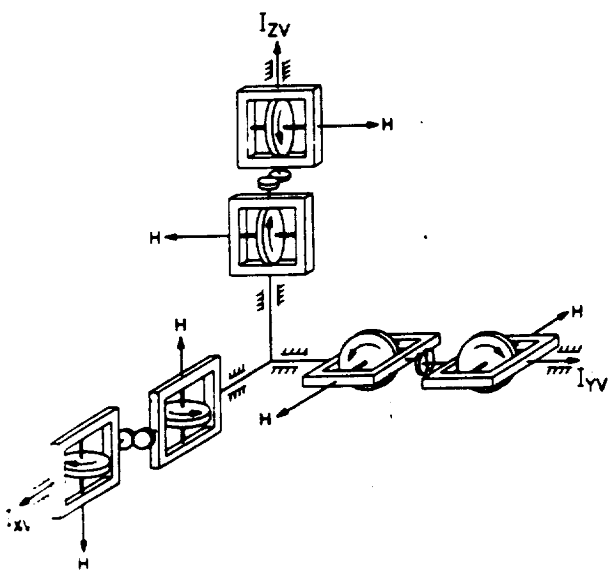
Klass, Philip J., "Firms Research Fiber-Optic Gyros As Successors to Ring-Laser Systems", Aviation Week & Space Technology, Feb. 13, 1989.

O'Connor, B. J., "A Description of the CMG and Its Application to Space Vehicle Control", J. Spacecraft and Rockets, Vol. 6, No. 3, March 1969.

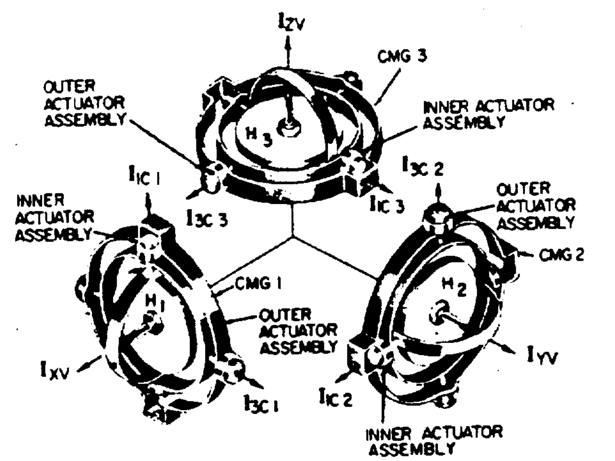
DIAGRAM 1*



a) Three single-gimbal CMG's



b) Three single-gimbal CMG pairs



c) The Sixpac configuration of 3 double-gimbal CMG's

* Taken from (O'Conner, 1969, p227)

DIAGRAM 2 *

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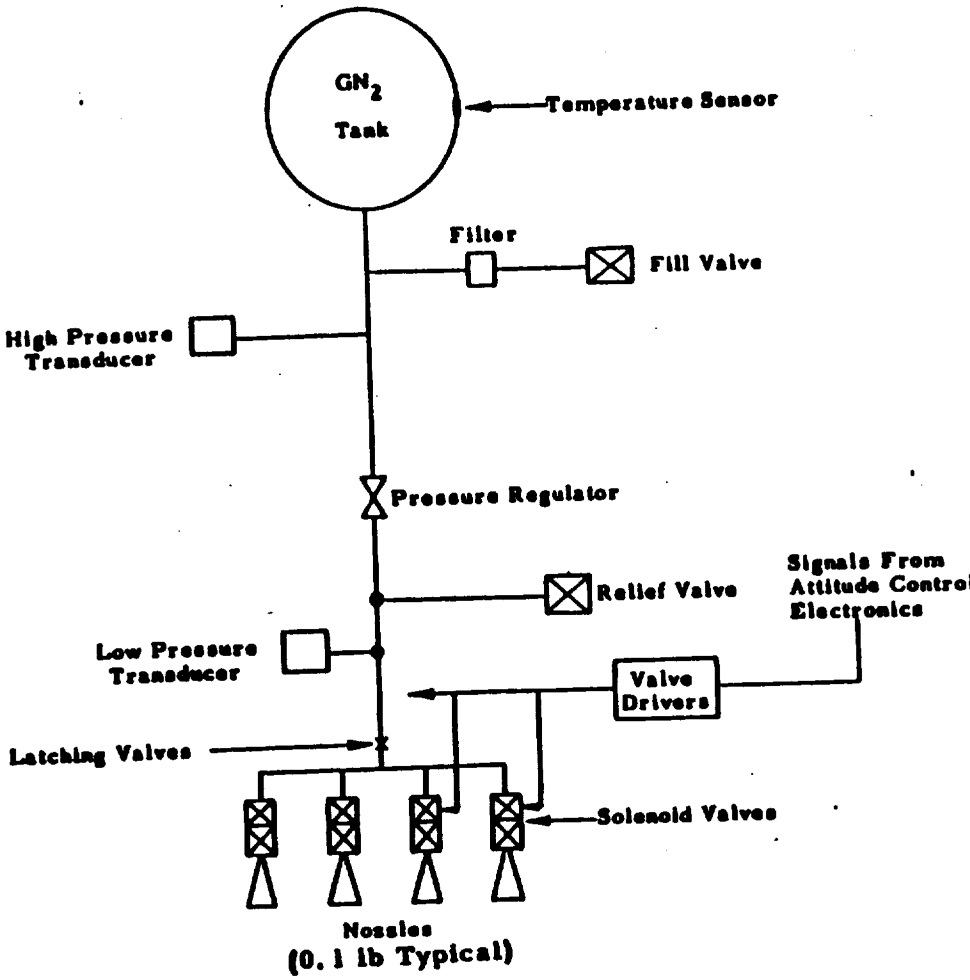

Cold gas (GN_2) Propulsion Subsystem

* Taken from (Chobotov, 1989, p77)

POWER SUBSYSTEM FOR ARC

GROUP 4

ROBERT BUENTE

INTRODUCTION

The driving facor in the Power Subsystem is reliability. In this way it differs from most of the other subsystems where weight and volume are primary concerns and reliability is assumed. This is due that in case of a power failure to any one of the subsystems the worst case scenario of loss of life becames not only a possibility but a foreseeable reality. The requirements of this subsystem support this idea. There is an extensive history of power use in spacecraft and this knowledge was drawn upon to conceive the final design.

STATEMENT OF REQUIREMENTS

The requirements, both stated and derived, are as follows:

1. Provide an uninterrupted source of power to spacecraft loads during mission life.
2. Protect main power bus and power units against damage due to load faults.
3. Protect user loads against outages and damage due to EPS unit failures.
4. Control and process power source and energy storage device outputs into forms compatible with other subsystem payload needs.

ANALYSIS OF PROBLEM

The major differentiating criteria for this subsystem are listed below in order of importance.

1. Reliability
2. Weight
3. Packaging
4. Cost

It is felt that reliability is the most important, by far, of these criteria.

The power needs by subsystem are listed below in Table I.

Table I - Power needs by subsystem

Subsystem	Power required
ECLSS	424 W
RRS	5 W
AAS	90 W
CDC	1700 W

This comes to a total power need of 2219 W. This figure was rounded up to insure that there would be enough power in case of remodeling. How is this power to be gotten? Solar cells alone cannot be used because of power

needs during both eclipse time and reentry. Batteries alone cannot be used because of the large weight they would require since there would be no recharging capabilities. The obvious choice is to use both solar arrays and batteries for this system.

There are two types of batteries that are projected to be available for 1994. These two are Nickel-Cadmium (Ni-Cd) and Nickel-Hydrogen (Ni-H). There are advantages and disadvantages to both types of batteries. These are stated in Table II.

Table II - Pros and cons of Ni-Cd and Ni-H batteries

Ni-Cd	Ni-H
advantages:	advantages:
- Extensive flight history	- No degradation of electrode
- Battery design exists	- Mass energy density greater than Ni-Cd
Disadvantages:	Disadvantages:
- Low specifc energy	- Some problems still exist
	- Higher cost not justified
	- Volume energy density less than Ni- Cd

Since the major driving factor in the power system is reliability, the battery choice is for Ni-Cd. Ni-Cd is also more cost efficient.

There are three major types of solar arrays. These are flexible blanket, planar rigid panel, and mini-concentration. The flexible blanket type array was chosen because it has the highest specific energy of the three.

After doing an analysis (AAE 241, Lec 13) that is in the appendix, the particulars of the power system could be found. These are presented in Table III.

Table III - Power system weight, volumes and areas

Component	Weight	Volume/area
Battery	85.2 kg	.0320 m^3
Solar array	84.0 kg	51.53 m^2

The solar array will be placed on a retractable beam so that it may be used on multiple missions. The process of positioning and placement was left to the Structures Subsystem.

The power bus chosen was the unregulated type. Table IV shows the advantages and disadvantages between the unregulated power bus and the regulated power bus.

Table IV - Pros and Cons between power bus types

Unregulated power bus	Regulated power bus
Advantages:	Advantages:
- Simplicity	- Constant stable supply voltage
- Low mass	
Disadvantages:	Disadvantages:
- Variations in voltage	- High mass
	- Less reliable

Again, the unregulated power bus was chosen only because it is a more reliable system than the regulated power bus.

It is important in the Power Subsystem that there be no single point failures. There are three common failure points. These are listed along with their solutions In Table V (AAE 241, Lect 13). All these safeguards will be implemented in the power system.

Table V - Common failures and resulting safeguards

Failure	Safeguard
Failure of EPS components	Fuse individual battery cells solar array strings
Failure in load components	Parallel redundant fuses on each load
Harness failure	Dual bus, diode operation of sources, double insulation of systems

The power to run ARC is not only going to come from its own power system. When it is docked with the space station it is assumed that all power needs will be met by the station. However, at all other times of operation, ARC's power system will have to carry the load.

CONCLUSION

Due to the magnitude of the power required, a rechargeable power system will be implemented. Ni-Cd battery cells and a flexible blanket array will be used to obtain this power. The design will feature a construction so that there are no single points of failure. The key driver in all the decisions for the Power Subsystem is reliability.

REFERENCES

Lembeck, Michael; "AAE 241 Lecture Notes", Kinko's, Lecture 13, Spring 1989.

APPENDIX

CALCULATIONS

These are the calculations to determine the size and weight required of the batteries and solar cells to produce the required amount of power. This procedure follows the method in the lecture notes for AAE 241. The following data is taken from Homework #12 in AAE 241.

Bus voltage, , 35 V

Maximum DOD (Ni-Cd), 45%

Energy density (Ni-Cd at 100% DOD)

Energy per battery cell (ni-Cd) 32 W-hr/cell

Solar cell efficiency at 25^{o} C 12%

(efficiency drop of 0.5% per ^{o}C)

Operating temperature (deployed array) 50^{o}C

Total degradation of solar cells (radiation, etc.) 30% in 5 years

Solar constant at 1.0 A.U. 1350W-m^2

Packing factor of solar cells 90%

The power load required is 2.3 kW. It is also stated that 30 minutes out of every 90 minutes are spent with the solar arrays shaded from the sun.

The number of cells needed are found by equation 1. The stored energy is found by equation 2, this needs to be known in order to solve 1.

$$\text{NO. CELLS} = \frac{\text{STORED ENERGY}}{\text{ENERGY PER BATTERY CELL}} \qquad (1)$$

$$\text{STORED ENERGY} = \frac{P_L T_E}{\text{DOD}} \qquad (2)$$

WHERE, P_L = POWER LOAD = 2300 W
T_E = TIME IN ECLIPSE MODE = .5 HOURS
DOD= DEPTH OF DISCHARGE = 45%

The stored energy is equal to 2556 W-h and the no. of cells required is 80. The battery capacity in amp-hours may be computed by means of equation 3.

$$C = \frac{P_L T_E}{\text{DOD x V}} \qquad (3)$$

WHERE V = BUS VOLTAGE = 35 V

The battery capacity is equal to 73 amp-h. The battery weight can be calculated by equation 4.

$$\text{BATTERY WEIGHT} = \frac{\text{STORED ENERGY}}{\text{ENERGY DENSITY}} = 85.2\text{ kg} \qquad (5)$$

The solar array analysis starts by computing the total power required to run the system and to charge the battery. This can be found by equation 6. First, however, the value for N must be found. This can be done by equation 7.

$$P_{BOL} = P_L + \frac{CV}{N} \qquad (6)$$

$$N < \frac{T_S}{\text{DOD}} \qquad (7)$$

WHERE T_S = TIME EXPOSED TO SUN = I HOUR

N is found to have a maximum value of 1.11. This gives the power required to be 4601.8 W. Then this number must be multiplied by 1 minus the degradation factor. This gives the value of 6574 W. The solar array area can be found from equation 8.

$$A = \frac{P_{BOL}}{S \times C_r \times e \times (1 - \alpha(T\text{-}25))} \qquad (8)$$

where, S = solar constant, 1 a.u.
C_r = packing factor, 90%
e = cell efficiency, 12%
α = temperature degradation factor, .5%
T = operating tmperature = 25°C

The solar array area is found to be 51.53 m^2. The weight can be found by multiplying this by the areal density of 1.63 kg/m^2. The weight of the solar array is 84 kg.

ONBOARD CHEMICAL PROPULSION SYSTEM FOR ARC

GROUP 4

BOB BUENTE

INTRODUCTION

The onboard chemical propulsion system has weight and volume as its driving factors for design. This system has benn designed to provide the needed Δv to propel ARC from the 100 mile drop off by the ELV to the space station, and then also to return ARC to Earth from the space station. It was deemed appropriate to use liquid oxygen and hydrazine as oxidizer and fuel, respectively. Please see Appendix A for all calculations.

STATEMENT OF SUBSYSTEM REQUIREMENTS

The requirements, both stated and derived, for the chemical propulsion subsystem are as follows:

1). Provide necessary Δv to reach space station and then return to Earth from said station.

2). Have the capability to return to Earth quickly in case of injury to space station crew.

3). ARC can not be accelerated faster than three g's at any time.

4). Provide safe, reliable operation.

5). Meet lifetime requirement of 5 years.

DESIGN CONSIDERATIONS, RESULTS AND PARTICULARS

The ELV will leave ARC in a 163 x 163 km orbit with an inclination of 28.5°. Space Station Freedom is in a circular orbit of 290 km also with an inclination of 28.5°. The Δv required for a Hohmann transfer to the space station from a 163 km orbit is .074 km/sec. The Δv required to return to Earth was calculated by the Reentry and Recovery Subsystem to be .14 km/sec. Therefore, the total Δv required by ARC is .214 km/sec. Table I shows the amount of fuel used and the time of each burn and also the final mass of the vehicle.

Table I - Burn schedule for required delta v's

burn number	initial mass	final mass	mass expelled	burn time
1	22,500 kg	21,859 kg	641 kg	54.7 sec
2	21,859 kg	20,697 kg	1162 kg	99.3 sec

The propellant/oxidizer choice is hydrazine and oxygen. Performance, weight and volume are the main drivers for fuel selection, followed by secondary considerations of toxicity and ease of usage. Table II is a chart of Dv for a fixed tank volume and given vehicle weight for some major fuel combinations. This chart combines both performance and volume data. The required mass of propellants is not going to differ greatly from fuel to fuel,

however the weight factor comes in as a func-tion of tank volume. The weight of the tank is proportional to the volume of the tank, i.e, the smaller the tank volume, the lighter the weight.

Table II - Delta v for a fixed tank volume and vehicle mass

In this way Table I takes the three main design drivers of fuel selection into consideration. From this point the propellants can be weeded out due to the secondary factors, such as toxicity and complexity of usage. Fluorine, for example, is very toxic and also is a corrosive when in contact with many materials. For similar reasons, nitrogen tetroxide must also be avoided. The choice of O_2/hydrazine was made because it displays good performance characteristics while harmful side effects are in an acceptable range. One of the advantages of hydrazine is that it is able to be used as a regenerative coolant for the thrust chamber. Some of the side effects and/or precautions to control them are the following:

1). Due to the low boiling point of liquid oxygen, all lines, tanks and valves that contain oxygen will have to be insulated to minimize evaporation.

2). Due to the high freezing point of hydrazine its contact materials must also be insulated.

3). Hydrazine is compatible with only a few metals, among these are stainless steels and 1100 and 3003 series of aluminum (Sutton,181).

This is by no means a perfect fuel. There are problems but it is felt that these problems are controllable when dealt with logically and carefully.

The design thrust was chosen to be 30,000 N and the chamber pressure was chosen to be 3.4475 MPa. Again, these were chosen to reduce weight and volume. Since engine volume and therefore weight is a function of thrust, it was necessary to keep thrust values low. Also, it was necessary to abide by the three g acceleration limit imposed by the system requirements. However, it must be admitted that 30,000 N, although it does meet these requirements, was merely a choice. The chamber wall thickness, and therefore the weight of the engine, is linearly proportional to chamber pressure. Here again 3.4475 MPa was chosen as a value within the acceptable limit.

The engine is made of stainless steel. This material was chosen for a combination of reasons. Some of these are high yield strength, good temperature conductivity, and ease of manufacturing. The first two reasons reduce the weight due to pressure and heat transfer aspects. The tanks were made of pressure vessel steel. The only other material that could have been used is aluminum due to the hydrazine corrosion factor. Pressure vessel steel has a lower ratio of density over yield strength than aluminum and thus was chosen.

CONCLUSION

The Propulsion System cannot be accurately designed by hand. There are a number of instabilities that will be noticed once in the development and testing stage. These will have to be corrected and is where the major amount of cost comes in. However, this design has a solid background.

REFERENCES

Sutton, George P., Rocket Propulsion Elements, Fifth edition, John Wiley and Sons, 1986.

Ashby and Jones, Engineering Materials 1, Pergamon Press, 1987.

APPENDIX A

CALCULATIONS

The velocity of a vehicle in a circular orbit about the Earth is given by equation (1).

$$v = \sqrt{\frac{\mu_o}{r}} \qquad (1)$$

From this equation the velocity of ARC at the ELV dropoff radius of163 km is found to equal 7.806 km/sec. The velocity of the space station at an altitude of 290 km is 7.732 km/sec. The Δv required for the maneuver from point 1 to point 2 is equal to $v_2 - v_1$ = .074 km/sec.

The method for this analysis can be found on pages 221-227 of reference [1]. The thrust of the engine was chosen to be 30,000 N. The chamber pressure was chosen to be 3.4475 MPa. The propellants were selected to be hydrazine and oxygen. The following values were then determined.

Propellants	hydrazine and oxygen
Chamber pressure, p_1	3.4475 Mpa (500 psi)
Thrust	30,000 N
Mixture ratio	.74
Chamber temperature, T_1	3027 ^oK
Mean molecular weight of exhaust gases	18.3 kg
Specific heat ratio	1.25

1. Propellant mass and expulsion rate

The velocity of the gases out of the nozzle exit can now be determined by equation (2):

$$v_2 = \sqrt{\frac{2k}{k-1}\frac{R'T_1}{M}\left[1-\left(\frac{p_2}{p_1}\right)^{(k-1/k)}\right]} \qquad (2).$$

This value, 2641 m/sec, is the ideal effective exhaust velocity. By using a correction factor of .97, the actual exhaust velocity is 2562 m/sec.

By using equation (3), known as Tsiolkovsky's equation, it is possible to find the total mass of propellant needed for the required Δv. It is assumed that the total wet mass of the vehicle will be the maximum allowed by the ELV, which is 22,500 kg.

$$m_p = m_o - \frac{m_o}{e^{\left(\frac{\Delta v}{v_2}\right)}} = 1803 \text{ kg} \qquad (3)$$

m can be found by F/v_2 and is equal to 11.71 kg/sec. Due to loss of propellant during ignition because of incomplete burning, 5 seconds worth of propellant will be added. This gives a final m_p equal to1860 kg.

2. Nozzle configuration

By using Figure 3-7 in [1], the nozzle coefficent C_F is found to be 1.45 while the nozzle area expansion ratio e is 5. The area of the throat, A_t, can be found from equation (4):

$$A_t = \frac{F}{C_F p_1} = .006 \text{ m}^2 \qquad (4)$$

A_e, the area of the nozzle exit, is equal to eA_t, and has a value of .03 m^2. From research, it seems to be a matter of course that the nozzle diffuser half angle be equal to 15^o.

3. Chamber configuration

A cylindrical shape was chosen for this chamber because it allows for simplicity of calculation of the diameters. Since the value of the chamber velocity is not readily calculated, it is assumed that it is 130m/sec. This is a reasonable assumption (Sutton, p.223). Knowing this, it is possible to estimate the cross-sectional area of the chamber. This is done by means of equation (5).

$$A_1 = \frac{F R' T_1}{v_2 \mathcal{M} p_1 v_1} = .036 \text{ m}^2 \qquad (5)$$

This gives a chamber diameter, d_1, of .214 m. The characteristic chamber length, L*, is the length the chamber had if it were a true cylinder and had no converging section. This value is typically between .8 and 3.0 m. A value of 2.5 m was chosen. Chamber volume is related to L* by equation (6). The converging angle of the chamber wall is 30^o.

$$V_c = L^* A_t = .015 \text{ m}^3 \qquad (6)$$

Since the greatest pressure is located in the chamber, the thickness that is necessary to insure against rupture there should be sufficient over the rest of the engine. The formula for wall thickness is equation (7).

$$t_w = \frac{p_1 r_1}{\sigma_y} \text{ x safety factor} \qquad (7)$$

The material chosen for the engine is stainless steel. It has a density of 7500 kg/m^3 and has a yield strength of 286 MN/m^2. A safety factor of three was chosen as sufficient for the propulsion system. Inserting these values into equation (7), the wall thickness is found to be .01m. Table II lists the dimensions, volumes, weights and center of mass for the engine.

Table II - Dimensions, volumes and weights of ARC engine

Throat area	.006 m^2
Throat diameter	.087 m

Exit area	.03 m^2
Exit diameter	.195 m
Nozzle diffuser half angle	15^o
Chamber volume	.015m^3
Chamber length	.461 m
Chamber converging angle	30^o
Wall thickness (uniform)	.01 m
Engine length	.663 m
Engine weight	26.44 kg
Engine centroid	.4305 m from nozzle exit

4. Injector design

A multiple hole injector was arbitrarily chosen for this system. It features 8 pairs of injection streams, each consisting of an oxidizer and a fuel stream. First it is necessary to find the mass flow of each propellant by equations (8) and (9).

$$\dot{m}_o = \frac{\dot{m}r}{r+1} = 4.98 \text{ kg/sec} \quad (8)$$

$$\dot{m}_f = \frac{\dot{m}}{r+1} = 6.73 \text{ kg/sec} \quad (9)$$

It is now possible to calculate the injector hole areas using a couple of assumptions. The first is that the pressure drop through the injector is 551.6 kPa. The second assumption is that both orifice discharge coefficients have a value of 0.75. The injector hole areas are found by equation (10). This formula gives the total area of each propellants

$$\sum A_p = \frac{\dot{m}_p}{C_d \sqrt{2\, \Delta p\, \rho_p}} \quad (10)$$

injector area. By dividing these numbers by eight, the individual injector areas are found. The velocity of the liquids as they exit the injector can be found by equation (11). The injection angles now need to be found so

$$v = C_d \sqrt{2 \, \Delta p / \rho} \quad (11)$$

that the resulting momentum will be in an axial direction. First assume that the oxidizer velocity has an inclination of 20^o. Then by the use of equation (12) it is possible to determine the angle of declination of the fuel stream.

$$\gamma_f = \sin^{-1}[r\,(\frac{v_o}{v_f})\sin\gamma_o] \quad (12)$$

Injector design parameter	fuel	oxidizer
flow	6.73 kg/sec	4.98 kg/sec
pressure drop in injector	551.6 kPa	551.6 kPa
injection velocity	25.11 m/sec	22.46 m/sec
# of injector holes	8	8
area of each hole	3.404 $\times 10^{-5}$ m^2	2.254 $\times 10^{-5}$ m^2

angle of hole w/nozzle axis +13.08° -20.0°

5. Heat transfer

The process for the calculation of heat transfer is filled with assumptions. This process will have to wait until the development and testing phase for particulars. The chamber and nozzle will be cooled through the regenerative method, using hydrazine as the coolant. The pressure loss through the coils can be estimated at 340 kPa.

6. Propellant storage tanks

The respective volumes of the propellants can be found by dividing their mass by volume. The propellants will be stored in spherical tanks made of a pressure vessel steel. Pressure vessel steel has a yield strength of 1000 MN/m^2 and a density of 7800 kg/m^3. The inner radius of the tanks can be found once the required volume is known. The thickness of these tanks to insure against rupture and leakage can be found from equation (14). A safety factor of three was considered to be sufficient.

$$t = \frac{p_p r_1}{\sigma_y} \text{ x safety factor} \qquad (13)$$

The volume of the tanks can now be calculated where $r_2 = r_1 + t$.

A gas pressure feed system will be used to expel the propellants from their storage tank. These tanks will contain air at a pressure of 16 MPa. A separate tank will be used for each propellant. The mass of air required for each propellant can be found by using equation (15).

$$m_o = \frac{p_p V_p k}{R\, T_o \left[1 - \left(\frac{p_p}{p_o}\right)\right]} \qquad (15)$$

The volume of air required can be found using the perfect gas law.

Tank purpose	pressure	volume	dry weight
oxygen storage	4.0 MPa	.720 m^3	210.6 kg
hydrazine storage	4.34 MPa	1.103 m^3	319.8 kg
air for oxygen	16.0 MPa	.392 m^3	404.9 kg
air for hydrazine	17.5 MPa	.603 m^3	689.9 kg

ORBITAL TRANSFER PROPULSION SUBSYSTEM

Steve Hermann

Introduction: The primary function of the propulsion subsystem is to provide the delta-v necessary for the logistics module to reach and dock with the space station, execute platform maneuvers, and to return back to earth. Subsequently, the total delta-v needed to meet these requirements is very large. In order for the propulsion subsystem to satisfy this large delta-v requirement it would have to be very large and very heavy which would create problems with launch vehicle constraints. Our solution to the problem is to have two separate propulsion subsystems which would split up these delta-v requirements. The first, an advanced chemical propulsion subsystem, will be fixed to the logistics module and will be used for delta-vs necessary for reaching the space station and for returning to earth. The second, a solar electric propulsion subsystem, will be located at the space station with the capability of being attached to the logistics module. This system will be used for various orbital transfers from the space station to orbiting platforms. Having this second system located at the space station minimizes the effects on mission planning. Preferably, this subsystem will be transported to the space station by the means of the space shuttle. By utilizing the shuttle our mass and volume constraints for the logistics module are not as limited.

In addition to the delta-v requirements the propulsion subsystems must be able to execute certain maneuvers within a specific time limit. The advanced chemical subsystem must be able to meet reentry time requirements for both emergency and scheduled returns. The electric propulsion subsystem must be able to perform the required platform maneuvers in a certain time constraint in case of an emergency at the space station; as the logistics module will be needed for emergency crew return.

Finally and most important is the protection of the crew and the space station. Certain safety precautions must be taken into account when designing the propulsion subsystems. For example, toxicity must be considered when selecting a propellant for the system and also whether the exhaust particles will contaminate either the space station or the logistics module.

This section of the report will concentrate on the development and design of the Orbital Transfer Propulsion Subsystem required to execute various platform maneuvers. The analysis of the advanced chemical propulsion subsystem is contained in another section of this report. See table of contents.

Component Selection:

Electric Thrusters: Three basic types of electric propulsion rockets were compared to determine which would be used for the Orbital Transfer Propulsion Subsystem. Two important performance parameters, specific impulse

and thrust-to-weight ratios were compared for Electrothermal, Electrostatic, and Electromagnetic Engines. Schematics of the engines along with a performance chart is included in Figure 1. The Electrostatic or Ion Engine was selected because of its high specific impulse and its technology status. It is with this type of thruster that the greatest improvement in performance has taken place over the past 10-15 years(1).

Typical working fluids for the Electrostatic Engine are mercury, cesium, and xenon. From the standpoint of thrust performance and cost, mercury is best suited. Mercury also has a high density which in turn requires small, lightweight tanks(2).

Power Supply: For an analysis of different power sources see the power section of the report. In order to provide the necessary power to the Electrostatic Engine different solar array configurations were analyzed. Performance parameters were compared for three different solar array types, a flexible blanket, a rigid panel, and a mini-concentrator. Figure 2 The flexible blanket type was selected because of its high specific power and relatively low array area. The two wing solar array will be supported by a retractable mast. An collection of Ni-Cd batteries will be used as an auxiliary power unit.

Supporting Structure: Now that a thruster and power subsystem have been selected a structure to contain these components and the Power Processing Unit is necessary. The supporting framework will be constructed of a Beryllium Lockalloy material. This lightweight yet strong material can be fabricated into headers, stringers, and panels for our subsystem(3).

System Proposal: The Electrostatic Propulsion subsystem, having a high specific impulse (4000- 20000 sec.), will have extremely low propellant mass requirements and a large delta-v capability. Most of these systems are used for interplanetary missions such as the Mariner Mark II, the Advanced Capability Explorer (ACE), and the Thousand Astronomical Unit Explorer (TAU)(4). The Orbital Transfer Propulsion Subsystem is designed to supply a delta-v much lower than the delta-v required for the previously mentioned missions. Our subsystem must provide the delta-v necessary to transport our logistics module to Platform 1 and to return it back to the space station. An delta-v analysis is included in Appendix I. The Orbital Transfer Propulsion Subsystem will consist of one vehicle capable of performing eight Platform 1 maneuvers. If the system is determined to be an effective and efficient means by which to execute the platform maneuvers a second system will be constructed.

The Orbital Transfer Subsystem will have a four engine ion propulsion subsystem which has the engines arranged in a clustered configuration around a central neutralizer subsystem. The neutralizer subsystem serves to neutralize the ion beams exiting the engine. A benefit of the clustered propulsion subsystem is that any number

of the 30cm Ion Engines can be used depending on the mission requirements. In addition, a spare neutralizer and auxiliary power supplies are included with this subsystem for redundancy (5).

The power source will be a deployable two wing, 13% BSF/BSR solar array with a Ni-Cd battery Auxiliary Power Unit. As a general guideline, an overall power-to-thrust ratio of 20 to 30 kW/N will be necessary(6). Our subsystem, with 1 engine operating, will supply a total thrust of about .9 Newtons which will require a power source of about 20 kW. An analysis of the solar array area and mass necessary to supply 20 kW of power is included in Appendix II along with a battery sizing analysis. For a summary of the system characteristics see Appendix III.

A diagram of the Orbital Transfer Propulsion Subsystem is shown in Figure 3.

Problem Areas: The major design issue with the Orbital Transfer Propulsion Subsystem is how it will be attached to the Logistics Module.

Some sort of adapting subsystem is needed both on the Logistics Module and the Orbital Transfer Subsystem. The design of this subsystem must attempt to minimize the effect on mission planning. The attachment process would ideally be executed autonomously. Ways to accomplish the attachment will have to be studied further.

Another problem is operating the thrusters near the space station. The exhaust plume of the Ion Thruster leaves the vicinity of the vehicle in a line of sight manner and should not create a problem unless the space station surface intercepts the exhaust plume(7). However, there is concern of mercury exhaust particles possibly contaminating the space station. The Logistics Module may need to be backed away from the space station and platforms by a resistojet before firing the thrusters.

A problem occurs when this subsystem is executing a platform maneuver with the Logistics Module and an accident on the space station occurs requiring an emergency crew return to earth. The Logistics Module must be returned to the space station to evacuate the crew, hopefully in time. Possible solutions to this problem must be looked into.

Conclusion: With the selection and integration of the components complete the remaining task is the optimization of the subsystem. A more detailed analysis of exactly how many thrusters will most effectively execute a particular platform change must be done along with an optimization of the solar array sizing. The Ion Thrusters, having a large delta-v capability, may be capable of transporting the Logistics Module to platforms far from the space station. Hopefully this subsystem will prove to be a most effective and efficient means by which to perform the required orbital transfers.

REFERENCES

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2.) Stuhlinger, E., "Solar Electric Propulsion for a Comet Nucleus Sample Return Mission," 38th Congress of the International Astronautical Federation, Oct. 10-17, 1987/Brighton.

3.) Agrawal, Brij N., Design of Geosynchronous Spacecraft, Prentice-Hall Inc., Englewood Cliffs, N.J., 1986.

4.) Aston, G., "Ion Propulsion Technology Requirements for Planetary Mission Applications," 18th International Electric Propulsion Conference, Sept.30-Oct.2, 1985, Alexandria, VA.

5.) Aston,G., "Ion Propulsion Technology Requirements for Planetary Mission Applications," 18th International Electric Propulsion Conference, Sept.30-Oct.2, 1985, Alexandria, VA.

6.) Loeb, H.W. and Bassner, H., "Solar and Nuclear Electric Propulsion for High Energy Orbits," 38th Congress of the International Astronautical Federation, Oct.10-17, 1987/Brighton.

7.) Deininger, W.D., "Electric Propulsion Produced Environments and Possible Interactions With the SP-100 Power System," 18th International Electric Propulsion Conference, Sept.30-Oct.2, 1985, Alexandria, VA.

Figure 1) Sutton, George P., Rocket Propulsion Elements 5th Edition, John Wiley & Sons Inc., 1986.

Figure 2) Lembeck, M., Class Notes 238.13.

APPENDIX I

Delta-v required to reach platform 1 and return back to space station :.114 km/sec

Space Station : inclined orbit of 28.5 degrees, altitude 290 km

Platform 1 : inclined orbit of 28.5 degrees, 330 by 430 km orbit

Hohmann Transfer (minimum energy)

delta-v total = 2(delta-v1+delta-v2) delta-v1 = .0116 km/sec.

delta-v2 = .0450 km/sec.

delta-v total = .1132 km/sec.

Propellant Tank Sizing:

Mass of Logistics Module at launch :	21000 kg
Mass of fuel burned to reach space station :	- 560 kg
Mass of 1/2 of payload :	-4000 kg

Mass of Module required to perform Platform 1 maneuver (Mi) 16440 kg

delta-v = gIspln(Mi/Mf) Isp = 5978 sec Mi = 16440 kg

delta-v = .1132 km/sec. This gives Mf = 16409 kg

Mp = Mi - Mf = 31.00 kg (Propellant necessary to execute maneuver)

Density of Hg = 13600 kg/m^3

Necessary tank volume for one Platform 1 maneuver : .00233 m^3

Necessary tank volume for eight Platform 1 maneuvers : .01900 m^3

For eight Platform 1 maneuvers :

Four propellant tanks each having radius = .1043 m

APPENDIX II

Power Analysis : 20 kw load, 30 min. out of 90 min. out of sun
Max DOD (Ni-Cd) 45%
Bus voltage (nominal) 35V
Energy density (Ni-Cd at 100% DOD) 30 w-hr/kg
Degradation time .7 hr
N 2.22 hr
Packing factor 90%
Solar cell efficiency 12%, efficiency drop .5%

Battery cells required :

No. of cells required = stored energy/watt-hrs./cell

stored energy = [Pl(Te)]/DOD = 22222 watt-hrs.

watt-hrs./cell = 32 for Ni-Cd battery

No. of cells = 695 Battery capacity in hours = stored energy = 22222 watt-hrs/35V= 634.9 amp-hrs

Battery weight = stored energy/watt-hr/kg= 22222 watt-hrs/30 watt-hr/kg = 740.73 kg

Solar Array Power required : Pbol = (Pl+CV/N)/Deg. time= 29143.5 watts

Solar Array Area : Area = Pbol/[SCre(1-alpha(T-25)) = 228.44 m^2

Solar Array Weight : (Array Areal Density 1.63 kg/m^2) Array Weight = 372.36 kg

APPENDIX III

Ion Propulsion Subsystem Characteristics

Ion Engine :

Input Power, kw	20
Specific Impulse, sec.	5978
Thrust, Newtons	.90
Beam Current, amp	7.0
Beam Voltage, volt	4409

Power Processor Unit :

Specific Mass, kg/kw	3.6
Input Power, kw	33.3
Lifetime, yrs.	8

System Specifications :

Input Power, kw (3 engines operating)	60
Total Thrust, Newtons	2.7
Total Mass, kg (includes solar array)	1115

- scaled values taken from AIAA paper 85-2000

Figure 1

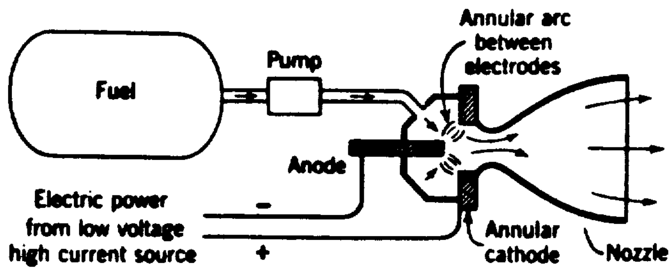


Fig. 1-8. Schematic diagram of arc-heating rocket engine.

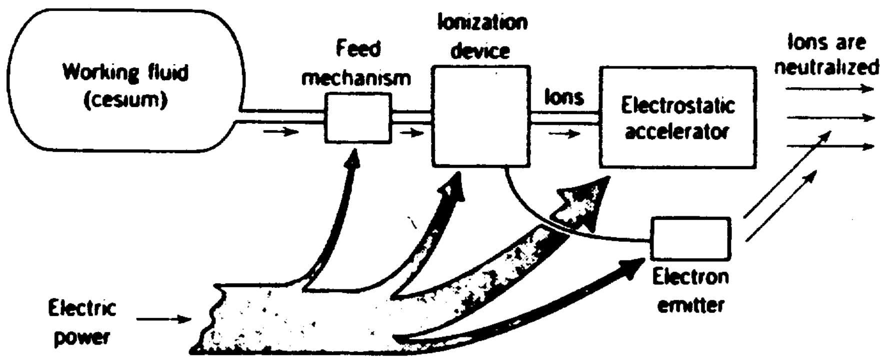


Fig. 1-9. Schematic diagram of a typical ion rocket.

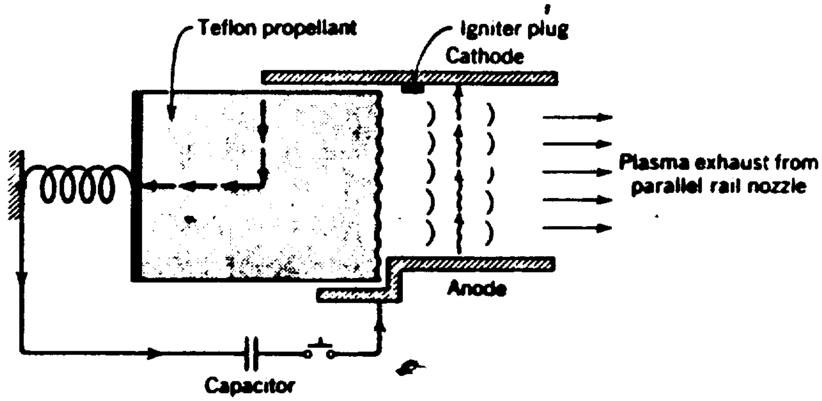


Fig. 1-10. Simplified diagram of a rail accelerator for a self-induced magnetic acceleration of a current carrying plasma.

SUTTON, p. 10, 11

Engine Type	Specific Impulse (sec)	Thrust to Weight Ratio	Typical Working Fluid
Electrothermal	400 to 2,000	10^{-4} to 10^{-2}	H_2
Electrostatic	4,000 to 25,000	10^{-5} to 10^{-3}	Cs
Electromagnetic	3,000 to 15,000	10^{-5} to 10^{-3}	H_2

Table values taken from Sutton, p.31 and Frisbee, p.450,451.

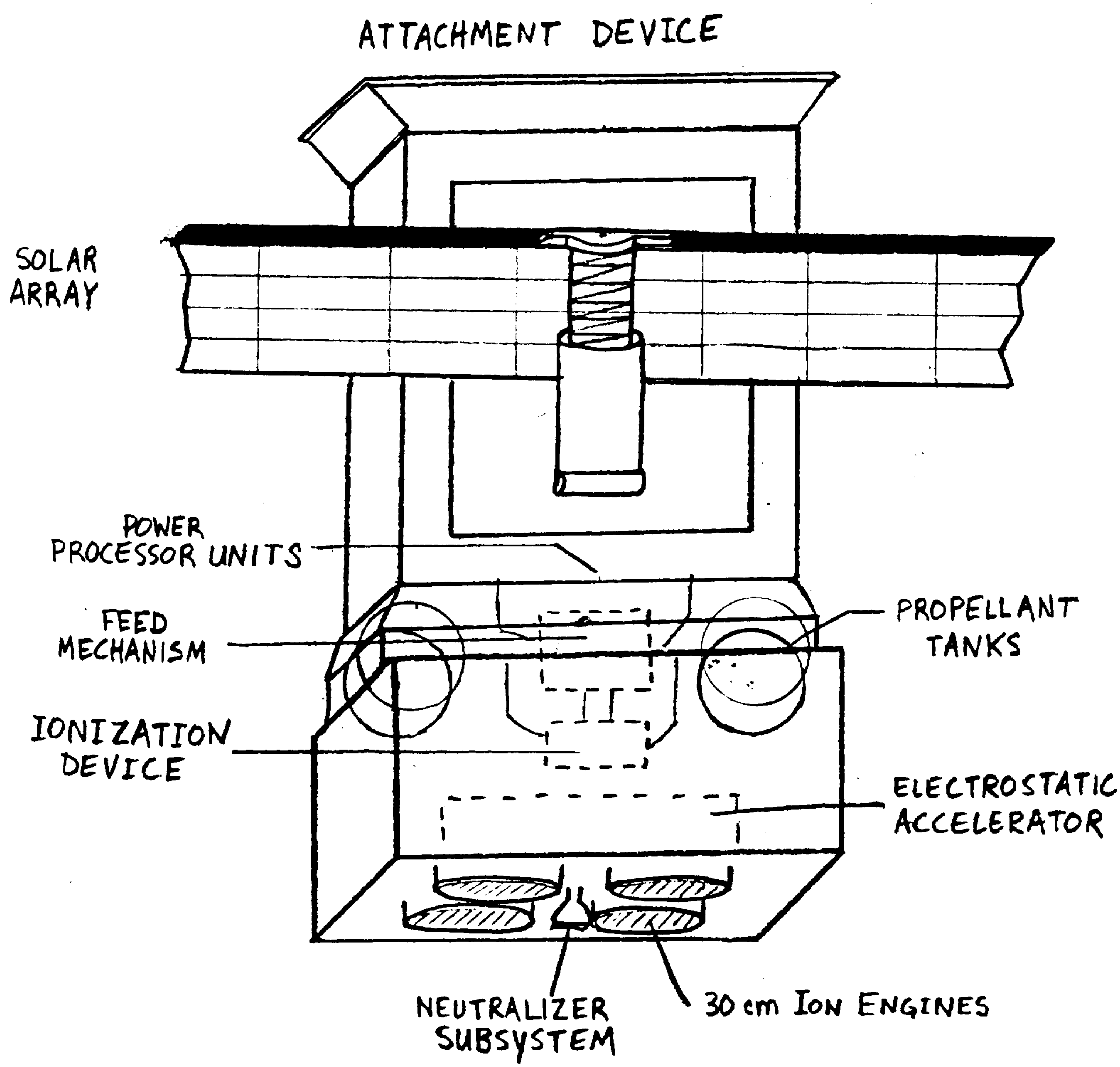
SOLAR ARRAY COMPARATIVE PERFORMANCE
50 kW (EOL) POWER, 10 YR MISSION

MISSION	ARRAY TYPE	CELL TYPE	CELL THK (MILS)	***FRONT SHIELD/ BACK SHIELD (MILS)	ARRAY AREA (2 WINGS) (M^2)	ARRAY WT (2 WINGS) (KG)	EOL POWER DENSITY ($WATTS/M^2$)	ARRAY AREAL DENSITY (KG/M^2)	EOL SPECIFIC POWER (WATTS/KG)
LEO 236 NMI i = 60°	FLEXIBLE BLANKET	13.3% BSR 13% BSF/BSR	8 2	4.5/6.5 4.5/6.5	516 500	944 815	96.8 100.0	1.83 1.63	52.8 61.5
	PLANAR RIGID PANEL	13.3% BSR 13% BSF/BSR	8 2	4.5/25 4.5/25	500 486	1420 1107	90.2 101.1	2.78 2.40	35.2 42.1
	MINI-CONCENTRATOR	20% GaAs	10	20/30 16/30	364 368	2193 1661	137.2 135.7	6.03* 4.51**	22.8 30.1
GEO 19300 NMI i = 0°	FLEXIBLE BLANKET	12% BSR 13% BSF/BSR	8 2	4.5/6.5 4.5/6.5	598 560	1076 880	83.6 89.3	1.80 1.58	46.4 56.8
	PLANAR RIGID PANEL	12% BSR 13% BSF/BSR	8 2	4.5/25 4.5/25	590 564	1650 1336	84.7 88.2	2.78 2.40	30.3 37.4
	MINI-CONCENTRATOR	20% GaAs	10	20/30 60/65	408 374	2463 1730	122.4 133.8	6.03* 4.63**	20.3 28.9
MEO 5600 NMI i = 60°	FLEXIBLE BLANKET	12% BSR 13% BSF/BSR	8 2	10.5/6.5 10.5/6.5	1106 1086	2280 1976	45.2 46.5	2.06 1.90	22.0 25.3
	PLANAR RIGID PANEL	12% BSR 13% BSF/BSR	8 2	17.5/25 17.5/25	741 693	2680 2242	67.5 72.1	3.62 3.23	18.6 22.3
	MINI-CONCENTRATOR	20% GaAs	10	20/30 96/87	687 392	4132 1860	72.8 127.7	6.03* 4.74**	12.1 26.9

*10 MIL NICKEL OPTICS; **5 MIL NICKEL OPTICS; ***EQUIVALENT FUSED SILICA

Figure 3

ORBITAL TRANSFER PROPULSION SUBSYSTEM



Attitude and Articulation Control

Comparison Studies

Attitude Control System:

	Advantages	Disadvantages
Cold Gas Jets	• can use next to space station • good for maneuvering	• weighty system • limited life • mechanically complex
Resisto Jets	• good for maneuvering • higher efficiency (I_{sp})	• weighty system • limited life • mechanically complex
Reaction Wheels	• mechanically simple	• large weight and power required • low accuracy
CMGs	• mechanically simple • low weight and power required • high accuracy possible	• requires complex control law • not suited for maneuvering

CMGs are the main system. Cold gas jets are used for docking and CMG unloading.

Sensor system:

	Advantages	Disadvantages
Horizon	• mechanically simple	• low accuracy
Sun	• simple design	• slow response
Stellar	• high accuracy	• slow response
Rate Integrated Gyros	• fast response • small size (R-FOG)	• requires use of three primary sensors

System will use 1 sun sensor, 2 star sensors and 2 sets of Resonant Fiber Optic Gyros (1 for redundancy).

= selected component

REENTRY AND RECOVERY SUBSYSTEM

PRIMARY DESIGN ISSUES

ISSUE	CHOICES	REASONS
BODY SHAPE	BLUNT OR **LIFTING BODY**	REDUCES G'S, INCREASE CROSSRANGE
THERMAL PROTECTION	COOLING, ABLATIVE **CERAMIC**	REUSABLE & SIMPLE, PROVEN TECHNOLOGY
DECELERATION	AEROBRAKING, RETRO-ROCKET, **PARACHUTE**	LOW WEIGHT, SIMPLE DESIGN
PARACHUTE TYPE	CONICAL,**RECTANGULAR**	MANEUVERABILITY, INCREASE CROSSRANGE
LANDING	WATER, **LAND**	NO WATER PROTECTION NEEDS, ACCESS TO TRANSPORTATION

POWER AND PROPULSION SUBSYSTEM

PRIMARY DESIGN ISSUES

ISSUES	CHOICES	REASONS
PROPELLANT	O_2/H_2, **O_2/HYDRAZINE**	LESS WEIGHT, ABLE REGENERATIVELY COOL CHAMBER
BATTERY TYPE	**Ni - Cd** or Ni -H	EXTENSIVE USE, DEVELOPED TECHNOLOGY

MISSION MANAGEMENT AND PLANNING SUBSYSTEM

PRIMARY DESIGN ISSUES

ISSUE	SELECTION	REASON
CREW OPTIONS/ VEHICLE NUMBER	2,4-MAN VEHICLES	OPTIMIZES LOGISTIC PAYLOAD AND COSTS
LAUNCH VEHICLE	TITAN IV	OPTIMIZES SUCCESS RATE, PAYLOAD CAPABILITY AND COST
TRAJECTORY OPTIONS	BALLISTIC PATH/ HOHMANN TRANSFER	MINIMUM ENERGY AND ΔV TRAJECTORIES
REQUIRED ΔV	REFER TO PROPULSION SYSTEMS	

COMMAND AND DATA CONTROL

GROUP #4

ANTENNA SYSTEMS	PRO	CON
Isotropic	• does not need to be aimed	• weaker signal gain
Parabolic Dish	• more signal gain	• must be aimed
Combined **SELECTED**	• has benefits of both • can perform more tasks	• requires more power

DOCKING SYSTEMS	PRO	CON
Microwave Interferometer	• low power • no scanning	• requires five antennas in basic configuration
Shuttle System **SELECTED**	• uses the present Ku antenna • research already accomplished	• antenna must be positioned

ECLS Subsystem Requirements and Summary of Equipment Choices

Environmental Control and Life Support Subsystem		
		Regenerative (closed-loop)
	*	Non-regenerative (open-loop)

O2, N2 storage	*	Pressure vesels
		Cryogenic storage tanks
CO2 removal	*	LiOH cannisters - no air revitalization
Potable H2O	*	Tank storage - no reclamation
Solid food		None stored on LRV - use SS stores
Required crew volume		Ample
Cabin atmosphere	*	Supplied by pressure vessel
Thermal control		Single-phase transport fluid w/ radiator
	*	Two-phase transport fluid w/ coldplate
Humidity control		Existing technology
Fire detection/suppression	*	Automatic - existing technology and
	*	smoke removal device
Medical equipment		None - provided on SS
	*	Seats fold down to a bed or stretcher support

* denotes equipment/subsystem used to fulfil ECLSS requirements

ARC

DESIGN LAYOUT

Group 4

Drawn by: Mark Mueller

The following abbreviations are used in this section for component referral to the various subsystems:

AACS	Attitude and Articulation Control Subsystem
CDCS	Command and Data Control Subsystem
ELCSS	Environmental Control and Life Support Subsystem
P&PS	Power and Propulsion Subsystem
R&RS	Reentry and Recovery Subsystem
Strc	Structure Subsystem

DESIGN LAYOUT

Center of Mass Calculation

The center of mass was assumed to be along the centroid of the ARC for simplicity. These calculations measure the center of mass from the tail of the craft. The cargo mass is approximated as 13000 kg evenly distributed in the cargo bay.

Item	Location (m)	Weight (kg)	Product
Nose structure	13.26	65.52	868.80
Cone structure	11.54	671.62	7750.49
Cylinder structure	6.24	2307.60	14399.42
Flange structure	0.54	580.76	313.61
Backwall	1.80	158.22	284.80
Bulkheads (12)	7.83	103.30	808.84
Floorboard	9.00	77.00	693.00
Robot arm	8.50	100.00	850.00
Robot track	7.14	50.00	357.00
Docking adaptor	8.50	313.00	2660.50
Carbon-carbon flap	0.25	109.00	27.25
Solar Array & arm	3.50	492.00	1722.00
Engine	0.44	26.44	11.63
Tanks for P&PS	0.77	3661.24	2819.15
Batteries	13.26	85.2	1129.75
ECLSS equipment	9.00	310.2	2791.80
CDC antennae	12.34	86.24	1064.20
CDC computers	13.16	87.30	1148.87
scan platform	8.50	70.00	595.00
R-FOG (1)	13.16	0.45	5.92
R-FOG (2)	1.40	0.45	0.63
CMG (1)	1.60	27.00	43.20
2 CMGs	6.00	54.00	324.00
reaction jets	7.03	0.10	0.70
N_2 tanks	0.77	33.60	25.87
parachute	6.00	1240.00	7440.00
cargo	7.30	13000.00	94900.00
		23,710.24	143,036.43

CM lies $\frac{143,036.43}{23710.24} = 6.03\,m$ from the tail

Note: ARC weight without cargo = 10,710.24 kg

Design Layout

Right Half Section

Components:

A. Batteries, Computers, R-FOG
 see P&PS, CDCS, AACS

B. Antennae
 see CDCS

C. ELCSS equipment
 see ELCSS

D. Scan platform
 see AACS

E. CMGs
 see AACS

F. Solar Array
 see P&PS, AACS, Strc

G. R-FOG
 see AACS

H. N_2 tanks for reaction jets
 see AACS

I. Propellant tanks
 see P&PS

J. Rocket Engine
 see P&PS

K. Robotic arm and track
 see AACS

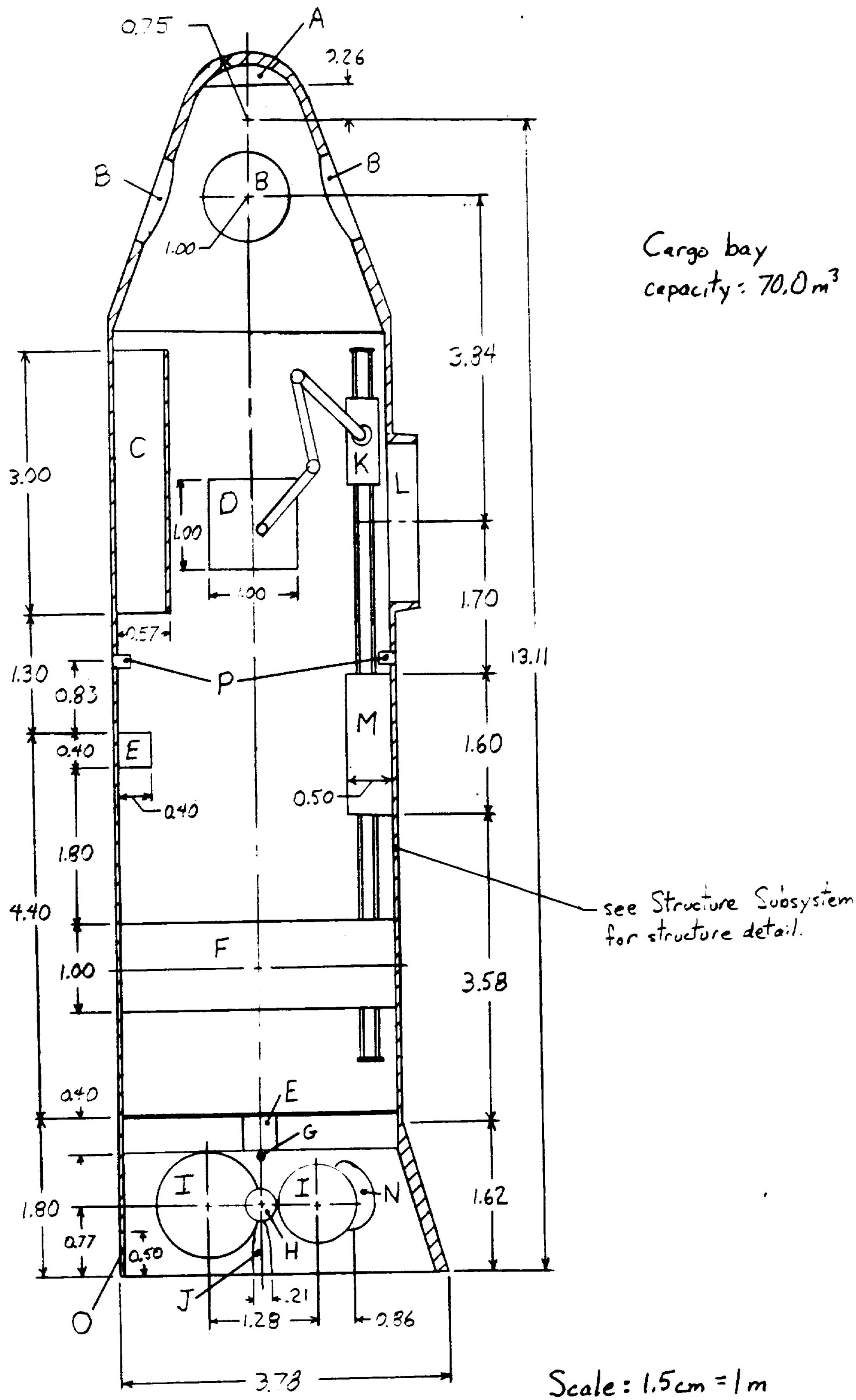
L. Docking adaptor
 see Strc

M. Guided Parachute
 see R&RS

N. Breaking Parachute
 see R&RS

O. Carbon-carbon flap
 see Strc

P. Reaction jets
 see AACS



Cargo bay capacity: 70.0 m^3

Scale: 1.5 cm = 1 m

Design Layout
Bottom Half Section

Components:

A. Batteries, Computers, R-FOG
see P&PS, CDCS, AACS

B. Antennae
see CDCS

C. ELCSS equipment
see ELCSS

D. Scan platform
see AACS

E. CMGs
see AACS

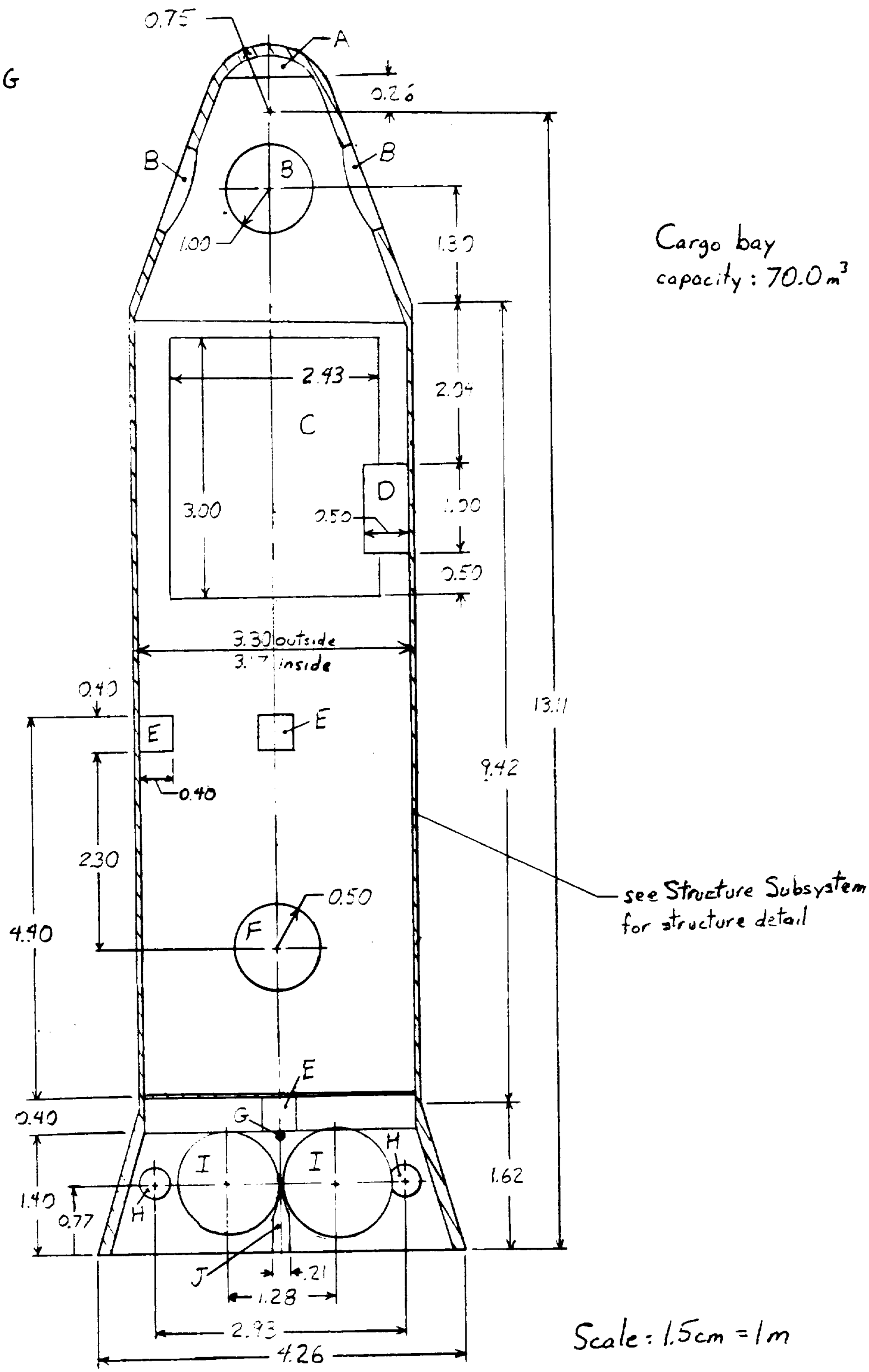
F. Solar Array
see P&PS, AACS, Strc

G. R-FOG
see AACS

H N_2 tanks for reaction jets
see AACS

I Propellant tanks
see P&PS

J. Rocket engine
see P&PS



Scale: 1.5cm = 1m

Multiple-Use Resupply and Personnel Habitat System (MURPHS)

Final Design Report

Submitted for AAE 241
Senior Design Project - Group Five
University of Illinois
May 1, 1989

This design project has been prepared by design group five. Each person listed below was responsible for a separate subsystem of the final module. In addition, everyone was responsible for partaking in several group decisions.

Mission Management and Planning

Chuck Martin, [redacted]

Structural Systems

Gene Wagner, [redacted]

Power and Propulsion Systems

Cliff Helfrich, [redacted]

Attitude and Articulation Control Systems

Beth Baird, [redacted]

Life Support and Crew Systems

Sonal Thakar, [redacted]

Reentry and Recovery Systems

Kevin Powers, [redacted]

Command and Data Control Systems

Steven G. Staats, Group Leader, [redacted]

Table of Contents

Introduction

The AAE 241 design groups were given the task of designing a logistic resupply and emergency crew return system for Space Station Freedom. This module is to be launched on an expendable launch vehicle currently in the U.S. inventory. It must be capable of carrying the maximum amount of supplies into orbit needed by the space station and other related orbiting modules during a ninety day base period. Once it has unloaded its supplies, it must be able to return waste to Earth from the station. In addition to this primary mission, the module must function as an emergency crew return system to bring astronauts back to earth from the space station.

The structure of the module will consist of three primary components: a logistic resupply capsule, a space station docking adapter, and an orbital transfer propulsion system. The module itself will have seven separate subsystems for the purposes of system integration: Mission Management, Planning, and Costing, Structural Systems, Power and Propulsion Systems, Attitude and Articulation Control Systems, Life Support and Crew Systems, Reentry and Recovery Systems, Command and Data Control Systems. Each of these individual subsystems will be covered separately in this report.

The design should allow the performance of different missions and carrying of several different payloads. It should also have a design lifetime that exceeds six years. The overall design should emphasize simplicity, reliability, low cost, and any advanced technology and artificial intelligence that are available before 1995 to allow for easier operation.

What follows in this report is Design Group Five's analysis of these requirements and the resulting system that is intended to fulfill this mission.

Mission Management, Planning and Costing

This is the mission management planning and costing subsystem of the M.U.R.P.H.'S. final design report. Included in this section are discussions and decisions made relating to the Request for Proposal which fall under the subsystem of Mission Management. Some of these requirements include the Launch Vehicle selection, the upmass and downmass, upvolume and downvolume, space shuttle use, mission timeline, and costing of the overall project.

In order that the Logistics Resupply Module or M.U.R.P.H.'S. make it into a low inclination earth orbit of 28.5° at a distance of 290 - 430 km, a launch vehicle must be selected to bring it into orbit. This vehicle must be capable of going into low inclination earth orbit and launching the M.U.R.P.H.'S. along with its resupply items and other subsystem components.

Choosing the launch vehicle was a relatively easy decision (see figure 1). To reach a low inclination orbit of 28.5° the launch must take place at Cape Canaveral, so this requirement rules out all Vandenberg launches. Early estimates of the upmass for all the subsystems were made and they totaled 27,946 kg. The only launch vehicle even close to this launch capacity is the Titan IV which can bring into orbit a maximum mass of 22,273 kg.

Going hand in hand with the launch vehicle selection is the selection of the number of vehicles to be used for a 90 day resupply schedule. Since the estimated mass for one vehicle is greater than the capability of the Titan IV, more than one vehicle must be used to bring up all resupply items. If two

MMPC Figure One: Launch Vehicle and Vehicle Number Selection

Subsystem	One Vehicle Up Mass (kg)	** ** Two Vehicles Up Mass (kg)
STRC	6,326	6,326
AACS	200	200
MMPC	16,220	8,110
RRS	500	500
PPS	2,300	2,300
LSCS	1,100	1,100
CDCS	1,300	1,300
Total mass	27,946	19,836

NOTE: MMPC mass based on 90 day resupple schedule

Titan IV Types

Vehicle Type	** ** 1	2	3	4
Orbit	100nm x 100nm	220nm x 220nm	100nm x 100nm	80nm x 445nm
Launch Site	CCAFS	CCAFS	VAFB	VAFB
Capability	22,273 kg	18,182 kg	17,995 kg	16,682 kg
Cost	$110 M	$110 M	$110 M	$110 M

NOTE: ** ** denotes the preferred selection

..M.

vehicles were used, the main factor in reducing the upmass would come from splitting up the resupply mass in half. If this is done (see Figure 1), the upmass is reduced to 19,836 kg which is a feasable weight to bring up in the Titan IV. So in the M.U.R.P.H.'S. system there will be two resupply vehicles bringing up supplies every ninety days.

There are a few variations of the Titan IV to choose from, but the obvious choice due to all the given requirements is vehicle type #1.

The M.U.R.P.H.'S. system will be launched by the Titan IV into a 100 nautical mile x 100 nautical mile or 184 km x 184 km low inclination orbit of 28.5°. It will orbit at a velocity V=(Ue/R) ** 1/2 where V=7.80 km/s. From this orbit the propulsion subsystem will perform an orbital transfer to bring it into the space station orbit where the attitude and articulation subsystem will dock it to the space station.

One of the main purposes of the M.U.R.P.H.'S. vehicle is that it needs to bring up all of the resupply items to the space station. These items vary in volume and weight, pressurized and unpressurized, frozen and room temperature, and rack/non-rack. An itemized list of the upmass and upvolume of the resupply items and of the other subsystems is shown in Figure 2. The resupply items' masses are half of the 90 day total. The totals of all the masses will then be the mass of one vehicle. In figure 3, another breakdown is given of 90 day logistics requirements, but these are also cut in half so that the totals will be for one

MMPC Figure Two: 45 Day Resupply Mass/Volume Summary

Resupply Needs	Upmass (kg)	Downmass (kg)	Up-volume (m^3)	Down-volume (m^3)
Crew:				
Food	735.7	--	3.950	--
Hygiene	65.4	--	1.232	--
Clothing	56.1	--	2.232	2.232
Station:				
Housekeeping	53.7	--	0.518	--
Waste Management	27.7	177.6	0.084	.0420
Trash	--	370.3	--	2.184
Spares	1139.6	1139.6	10.143	10.143
ECLSS Fluids	180.7	0	0.434	0
EVA Support	250.9	250.9	0.694	0.694
Customer:				
MTL	1091.4	1002	5.02	4.781
Plant/Animal	524.8	524.8	4.165	4.151
ESA Research	942	942	4.844	4.844
Customer Servicing	264.2	0	.0523	0
Human Research	16.1	23.7	0.056	0.056
Japanese	270.5	251.4	2.072	2.044
Other Subsystems:				
AACS	200	200		
PPS	2300	1753		
LSCS	1100	1100		
STRC	6,326	6,326		
RRS	500	500		
CDCS	1300	1300		
Totals	19,836	17,826	34.53	31.30

Resupply and waste totals only

:M

MMPC Figure Three: Forty-Five Day Total Logistic Requirments for One Vehicle

	Pressurized		Unpressurized		Fluids		Propellants	
	Crew/Sta	Customer	Crew/Sta	Customer	Crew/Sta	Customer	Crew/Sta	Customer
Mass Up (kg)	2074.3	2477	265.5	2076.1	180.3	182.6	22.7	840.9
Mass Dwn (kg)	1749	2378.7	265.5	2076.1	0	86.9	0	0
Vol. Up (m^3)	7.39	6.96	2.27	16.32	0.235	0.25	0.285	0.84
Vol. Down (m^3)	5.75	6.88	2.27	16.37	0	0.085	0	0

	Totals
Mass Up (kg)	8110.5
Mass Down (kg)	6547.2
Volume Up (m^3)	34.53
Volume down (m^3)	31.30

vehicle. So, in actuality, one vehicle is going to bring up a 45 day supply of items to the space station.

The upvolume shown in figures 2 and 3 has a large effect on the sizing of the cargo areas for these resupply items. Each of the two vehicles will take up equal amounts of the same items, thus, at least 34.53 m^3 must be alloted in the cargo area. Our cargo area will be 70 m^3. This will allow for extra cargo to be brought up and back at necessary times along the mission timeline which will be discussed later, and will also allow for added volumes of tanks, containers, loss of volume due to circular curvature of vehicle, and room for removal. Some freezers will also be brought up in the cargo area to be used to keep experiment samples which need to be frozen on the way down.

Some of the items which come up and down need to be pressurized. In order that this be possible, one half of the cargo area or 35 m^3 will be pressurized with air at standard temperature and pressure. This will be done very similar to the way the life support subsystem pressurized the crew cabin. Using the ideal gas law of PV = mRT and using STP values and a volume of 35 m, the mass of air needed is 41.96 kg, but this is only for the up pressurization so 83.92 kg will be needed for the whole cycle. This air will be put in stainless steel tanks with a thickness of .134 m, and a radius and height of .381 m. These values were obtained using equations from the life support subsystem. The cargo chamber will also be heat controlled by a system similar to the one for the crew members only it will be larger to accomodate the larger volume.

Since the upmass and upvolume are larger than the downmass and downvolume, they were the major design criteria for the size of the cargo area. But the down items are still an important part of the overall system. Almost every item that comes up also comes down, but some upmass is used up which makes the downmass and volume less. Figures 2 and 3 show the downmass and volume in the same way as it did the upmass and volume, with the exception that the other subsystems' downvolume and upvolume are not given because they will not take up space in the cargo area.

In bringing the items to the space station and back there needs to be an organized system inside the cargo area. Containers will be needed for some items, these will be of varying shapes and sizes as needed but will not exceed a 1.2 m width or height so that it will fit through the hatch. Racks will also be put in for items to placed in so that movement inside the cargo area is non-existent. The final placement of the racks, containers, and other items will be determined by the structure's subsystem.

Figures 2 and 3 only show the mass and volume totals for 90 day intervals. But every year and every two years these mass and volume totals are markedly larger than the 90 day interval (see figures 4 and 5). They show that every fourth and eighth interval more mass and volume needs to go up and come down. Every fourth interval this increase is about 300 kg and 7.35 m^3. Every eighth interval this increase is about 3200 kg and 14.21 m^3. The extra unused volume on the cargo area will be able to handle these additions, but this fact needs to be pointed out because

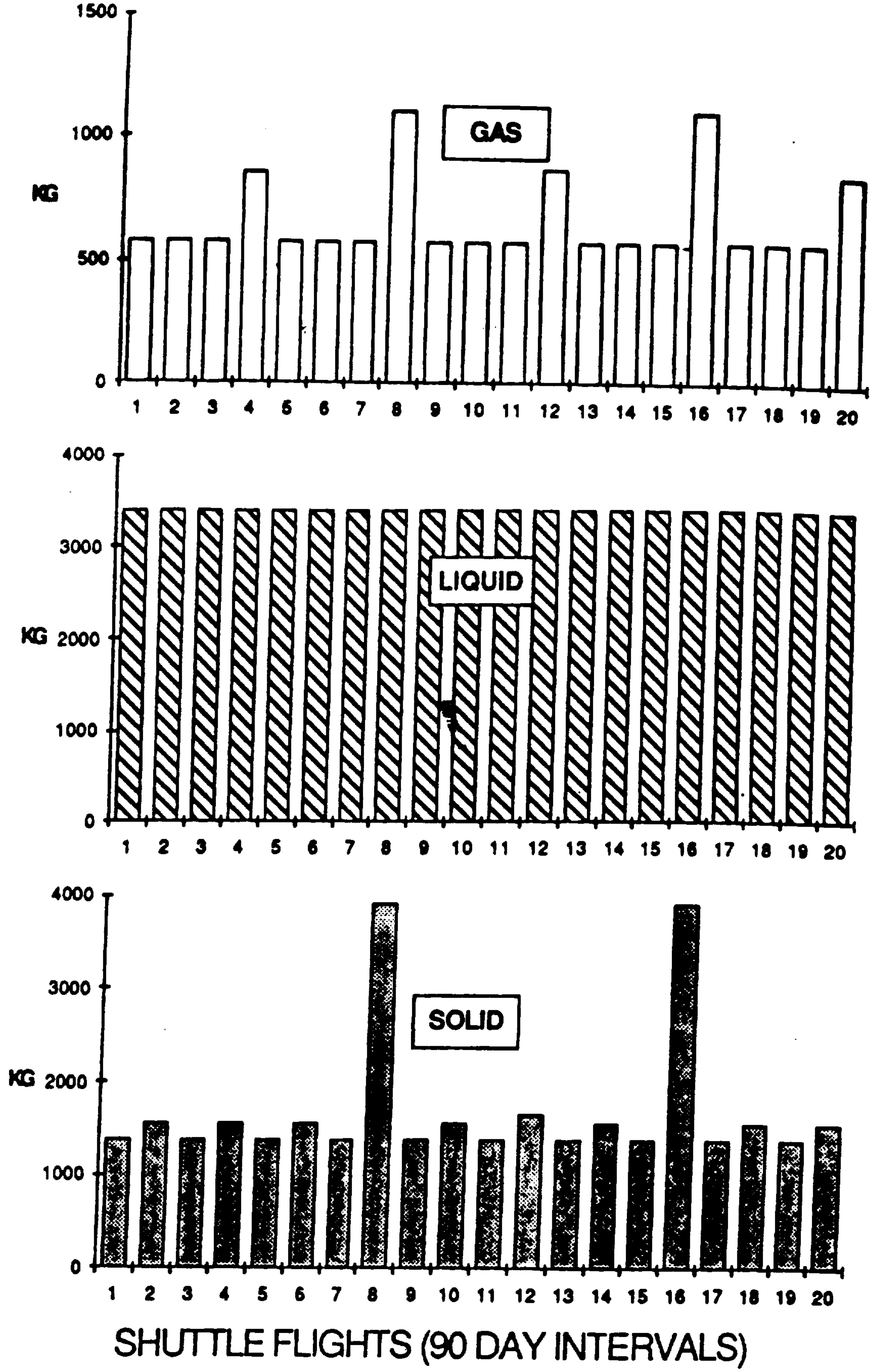


FIGURE 3. TOTAL OSSA MISSION WASTE PER 90 DAY PERIOD (MASS)

MMPC Figure 4

11

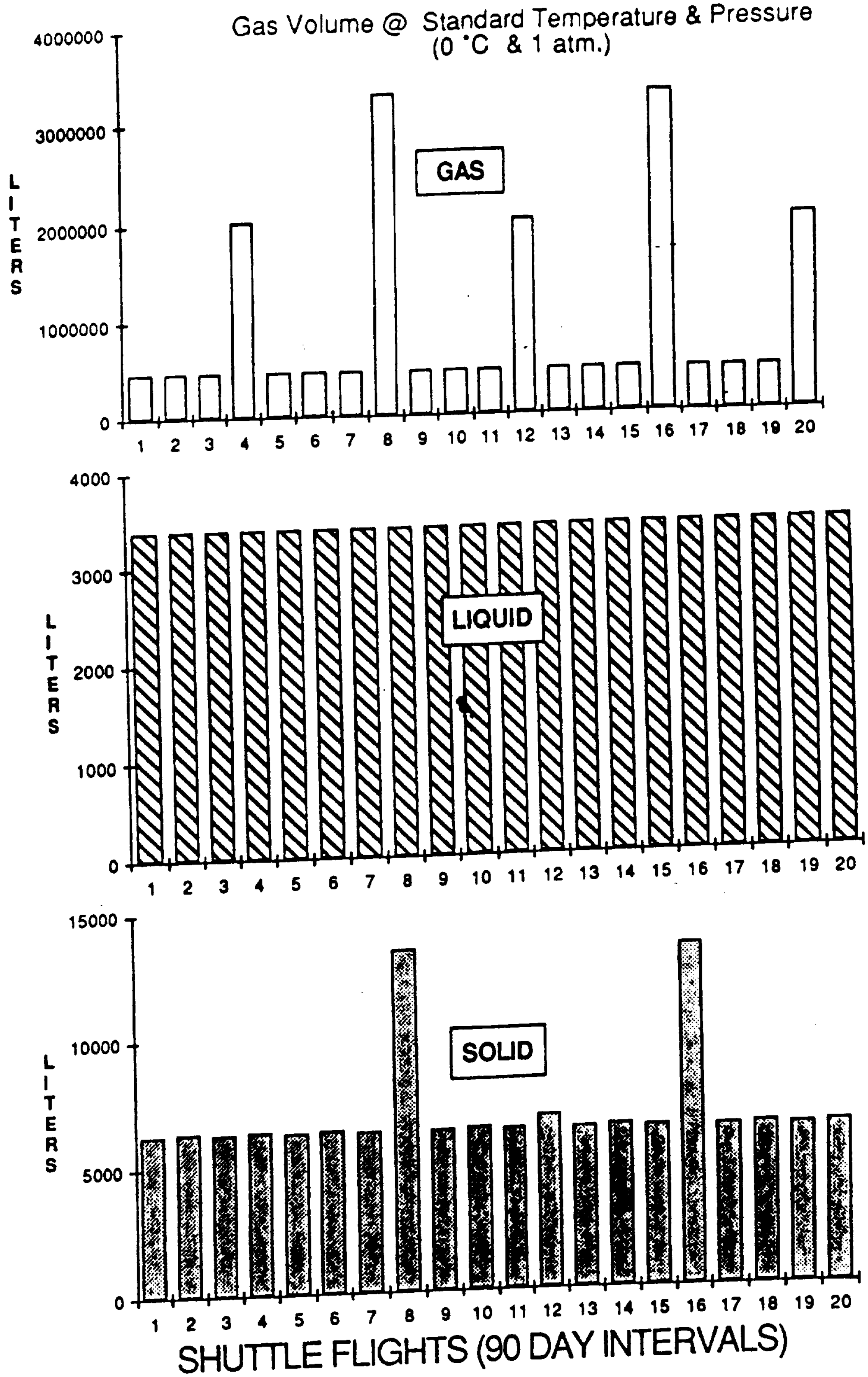


FIGURE 4. TOTAL OSSA MISSION WASTE PER 90 DAY PERIOD (VOLUME)

m MMPC Figure 5

some things will change with these additions, but since they only happen once every year and once every two years these changes will not be discussed in this report.

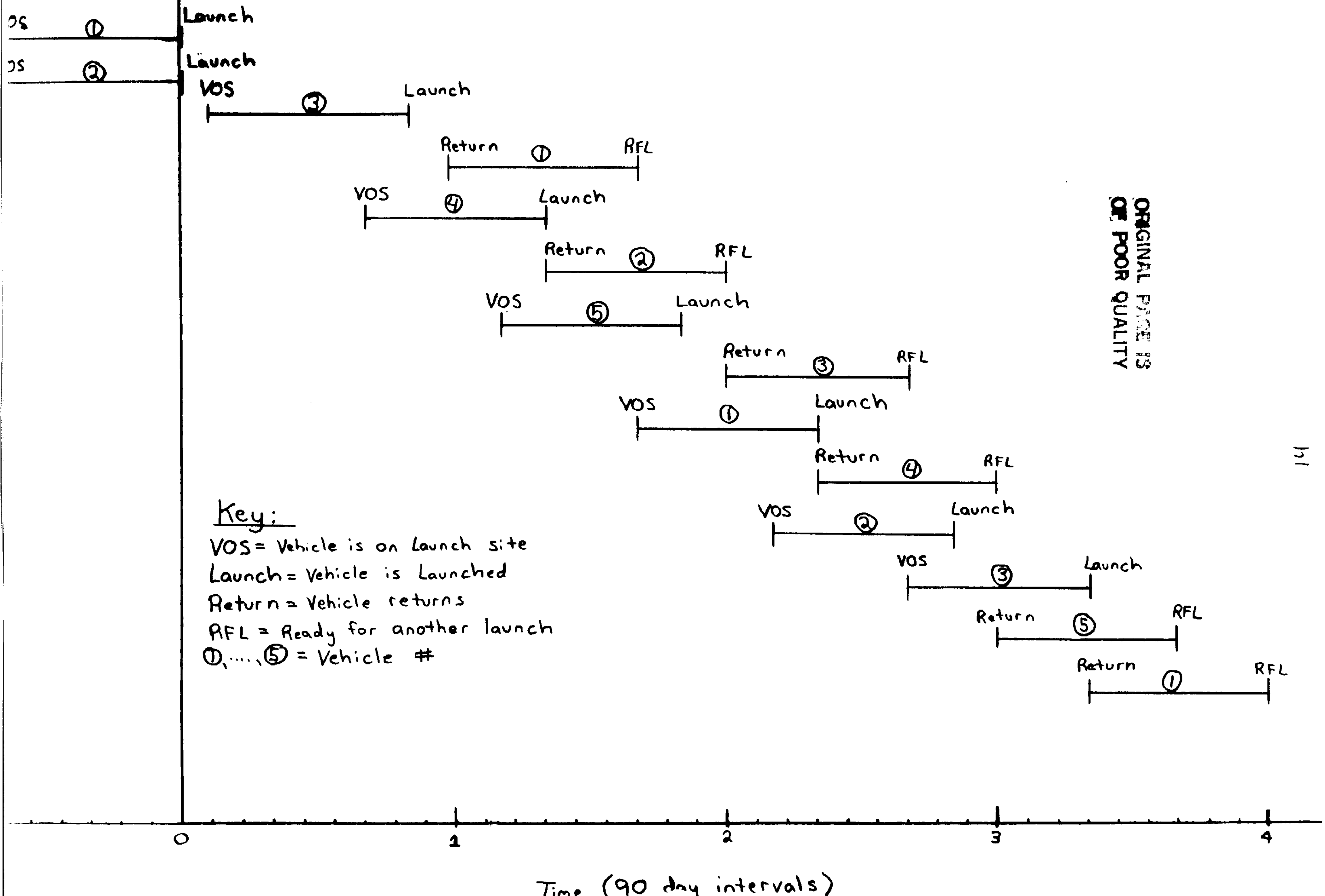
Now that what is going to be inside the vehicle has been determined, a mission schedule can be worked out and the number of vehicles in the overall system can be determined. The main factors influencing this timeline are the vehicle turnaround time, and the launch site schedule.

Although its hard to determine turnaround time without knowing what will happen each mission, we have estimated a turnaround time of sixty days for the M.U.R.P.H.'S. vehicle. Factors influencing this choice are: possible damage on mission, cleaning and unloading, inspections, transport, fatigue, resupply or replace internal systems, reloading, and other unseen problems. Sixty days is only an estimate, but it will be our assumption in making out the mission timeline.

The launch site schedule for the Titan IV is a total of 150 shifts, each shift consisting of eight hours. This means that on the pad a launch could take from as short as fifty days to as long as 150 days.

Keeping these two factors in mind, a mission timeline is prepared and can be seen in figure 6. The first two launches are done at a time zero. The timeline shows that there will always be at least two vehicles docked to the space station at all times, and sometimes three. There always has to be two vehicles docked in case of an emergency evacuation. Three vehicles will be docked for the last fifteen days of every ninety day period,

Mission Timeline



this will aid in the loading and unloading of the waste and supplies.

The launches are staggered at forty-five and sixty day intervals. This is to allow enough time in case of problems on the launch pad, at the space station, weather, or some other unseen difficulty causing a delay in the launch or return of a M.U.R.P.H.'S. vehicle. New supplies arrive fifteen days early on all resupply missions, and on each resupply, forty-five days worth of supplies is included. The return of each vehicle full of the waste occurs thirty days apart; this occurs due to the early launching of the resupplies needed and the fact that there always has to be two vehicles docked.

This timeline could be changed, but as it stands, using the stagger of the launches allows for there to only be five vehicles in the system. If there was no stagger, six vehicles would be needed and this would add to the overall cost of the system.

One of the requirements in the RFP states that the LRV has to be able to fit into the space shuttle cargo bay. It is assumed that this is for a return of the M.U.R.P.H.'S. vehicle if it is not capable of re-entry. The shuttle's cargo bay is 60 ft. long x 15 ft. in diameter, while the M.U.R.P.H.'S. vehicle is 44 ft. long and 12 ft. in diameter, so the vehicle can fit in the shuttle's bay. But the shuttle can only bring down 13,636.4 kg and M.U.R.P.H.'S. downmass is 17,826 kg. This makes that requirement infeaseable unless the waste downmass in the M.U.R.P.H.'S. vehicle is removed. If this was done, the vehicle

would only weigh 11,309 kg, and then the return in the shuttle would be possible.

The space shuttle can also be used for support if needed, but the M.U.R.P.H.'S. vehicle meets all the necessary requirements so it will not be used unless there is an emergency.

Another RFP requirement is costing the overall system. This is a difficult requirement to meet with any kind of accuracy. Only certain subsystems have a good estimate of their costs and others have none. The formula given to cost subsystems is very ambiguous, and due to development and technology and some of the subsystems using items not developed fully yet, the overall cost of the whole M.U.R.P.H.'S. vehicle is unknown. Although when design decisions were made the best systems were chosen with cost being one of the major considerations.

To conclude, this subsystem of Mission Management, a review of all the requirements will be given. Upmass and volume, downmass and volume were discussed and how they affected the design decisions. A launch vehicle was selected to fit these masses and volumes. A timeline of missions is shown for a one year period, along with the overall number of vehicles to be used. And the space shuttle's input was discussed. The other subsystems will define and answer the requirements of their subsystems and then the project will be complete.

MURPHS - STRUCTURES SUBSYSTEM

The most obvious challenge presented to the structures subsystem is the design of a vehicle that will be carried into orbit by a launch vehicle in the current United States inventory. For MURPHS, the answer is the Titan IV, which accomodates an approximately 16.77 meter by 5.08 meter cylinder. As can be seen by the schematics of MURPHS referenced a little later, this requirement is easily fulfilled once the payload is distributed between two modules.

Of primary concern to MURPHS is the selection of a material to effectively combat the space environment and survive a reentry into the Earth's atmosphere. The material used in the structure must also be able to withstand impacts by micrometeroids less than four inches in diameter. Those greater than four inches are tracked and can be corrected for. In addition, the material must be resistant to excessive radiation from space. After comparing many materials, it was narrowed down to the aluminum alloys, the titanium alloys, and a carbon-carbon composite. The carbon-carbon composite emerged as the most obvious choicedue to its low density and extremely high Young's Modulus (STRC Fig. 1). The carbon-carbon composite is fabricated by weaving strong carbon fibers into a two-dimensional mesh, similar to most composites. The mesh, resembling a fabric, is then saturated with a resin and heated to form the fiber/matrix system. After oxidation

MATERIAL	DENSITY (lb./in.^{3})	WEIGHT (lb.)	YOUNG'S MODULUS (PSI)	COST/LB (1974 $)	TOTAL COST (1974 $)
Al Alloy (24S-T)	.1	24.30E3	10.5E6	1.64	39.85E3
Titanium Alloy	.164	39.85E3	16.0E6	11.00	438.4E3
Carbon-Carbon Composite	.06	14.58E3	44.0E6	40.00	583.2E3

STRC Figure 1. Candidate Materials for Structure of Module.

protection is completed by applying a coating, the material is easily capable of maintaining its strength and thermal protection up to temperatures greater than 4000 F (STRC Ref. 1). This material has been used in aircraft brakes, rocket motors, missles, and spacecraft. It is used on the nosecone of the Space Shuttle and on its leading edges of the wings. Advatages of the carbon-carbon composite include high strength/density, high modulus/density, low density (.06 lb./in^3), increasing strength to 4000 F, excellent formability, good thermal and electrical conductivity, very low thermal expansion, and no melting point (STRC Ref. 2). It also will not contaminate optical surfaces due to the fact that it does not outgas.

In order for MURPHS to avoid serious micrometeroid damage, it will be fabricated with dual walls of the carbon-carbon composite described above. The outer wall will be .6352 cm (¼ in.) thick and the outer wall will be 1.270 (½ in.) thick. This is enough to avoid serious damage to the inner wall, while giving thermal protection also. Approximately two-thirds of the entire module is covered by this material in this configuration. The bottom third is enhanced by added insulation and is further discussed in the reentry section. Since the composite has such a high Young's Modulus and is thermally sound to 4000 F, this design will be safe and light.

One item needed to be taken into consideration is the fact that the carbon-carbon structure is a blackbody and would tend to absorb radiation. Thus the module needs a coating that demonstrates a low solar absorptance and a high thermal emittance.

Solar absorptance is typically the predominant external heat input to a spacecraft, whereas thermal emittance controls the rate at which heat leaves the spacecraft. STRC Figure 2 shows a variety of coatings with their respective properties. Magnesium Oxide White Paint was chosen because of its excellent properties, along with its lighter weight. (STRC Ref. 3)

The total mass of the structure and total inertia tensor is shown in STRC Fig. 3. This figure is derived from the INERT program on the IBM AT's. Each of the main components are listed separately by inertia tensor ($kg\text{-}m^2$), center of mass (x,y,z; from middle of the payload area, and mass (kg). The total center of mass needs to be a little aft and below the center of the ship (0,6,0). This is to insure that the ship will keep a nose-high attitude during reentry. It cannot be to drastic though or it will cause problems when parachuting down to Earth and trying to land straight up. The center of mass of the payload section can be placed almost anywhere by a skilled load master such as those on a C-5 crew. In this way, the centroid can always be kept in the same place by moving the payload around. STRC Fig. 4 shows the drawing of the module with the placement of the main components.

The four retro rockets are placed on a platform so that they can swivel in any direction for course correction during landing and to slow it before contacting the ground. Also on landing, retractable gear like that employed on the Apollo Lunar Module will support the weight of the ship on the lakebed it lands on. These legs will protect the rockets from ground collision.

WHITE COATINGS

	$\bar{\alpha}_s$	$\bar{\epsilon}_n$
Barium Sulphate with Polyvinyl Alcohol	0.06	0.88
Biphenyl–White Solid	0.23	0.86
Catalac White Paint	0.24	0.90
Dupont Lucite Acrylic Lacquer	0.35	0.90
Dow Corning White Paint DC-007	0.19	0.88
GSFC White Paint NS43-C	0.20	0.92
GSFC White Paint NS44-B	0.34	0.91
GSFC White Paint MS-74	0.17	0.92
GSFC White Paint NS-37	0.36	0.91
Hughson White Paint A-276	0.26	0.88
Hughson White Paint A-276 + 1036 ESH UV	0.44	0.88
Hughson White Paint V-200	0.26	0.89
Hughson White Paint Z-202	0.25	0.87
Hughson White Paint Z-202 + 1000 ESH UV	0.40	0.87
Hughson White Paint Z-255	0.25	0.89
Mautz White House Paint	0.30	0.90
3M-401 White Paint	0.25	0.91
Magnesium Oxide White Paint	0.09	0.90
Magnesium Oxide Aluminium Oxide Paint	0.09	0.92
Opal Glass	0.28	0.87
OSO-H White Paint 63W	0.27	0.83
P764-1A White Paint	0.23	0.92
Potassium Fluorotitanate White Paint	0.15	0.88
Sherwin Williams White Paint (A8W11)	0.28	0.87
Sherwin Williams White Paint (F8W2030)	0.39	0.82
Sherwin Williams F8W2030 with Polasol V6V241	0.36	0.87
Sperex White Paint	0.34	0.85
Tedlar White Plastic	0.39	0.87
Titanium Oxide White Paint with Methyl Silicone	0.20	0.90
Titanium Oxide White Paint with Potassium Silicate	0.17	0.92
Zerlauts S-13G White Paint	0.20	0.90
Zerlauts Z-93 White Paint	0.17	0.92
Zinc Orthotitanate with Potassium Silicate	0.13	0.92
Zinc Oxide with Sodium Silicate	0.15	0.92
Zirconium Oxide with 650 Glass Resin	0.23	0.88

STRC Figure 2. Candidate Coatings for MURPHS Outer Structure.

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[illegible] SPACECRAFT INERTIA RESOLVER V1.[illegible]
Mon May 01 19:55:26 1989
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ORIGINAL PAGE IS
OF POOR QUALITY

```
ADDING: outershell INERTIA TENSOR

      58560.0000        0.0000          0.0000
          0.0000     15210.0000          0.0000
          0.0000         0.0000      55560.0000

outershell CM:    -3.6070          0.0000          0.0000
outershell MASS: 6[illegible]

ADDING: pa[illegible] INERTIA TENSOR

      [illegible]                [illegible]                [illegible]
          [illegible]     13300.[illegible]          [illegible]
          [illegible]         [illegible]      [illegible]

[illegible]          1.[illegible]          [illegible]
[illegible]

ADDING: cabin INERTIA TENSOR

      1466.0000        0.0000          0.0000
          0.0000     1840.0000          0.0000
          0.0000         0.0000      1466.0000

cabin CM:    -3.6070          4.5730          0.0000
cabin MASS: 1100.0000

ADDING: nose/chute INERTIA TENSOR

      1118.0000        0.0000          0.0000
          0.0000     1724.0000          0.0000
          0.0000         0.0000      1118.0000

nose/chute CM:    0.0000          7.3930          0.0000
nose/chute MASS: 1282.0000

ADDING: instuments INERTIA TENSOR

      1175.0000        0.0000          0.0000
          0.0000     2175.0000          0.0000
          0.0000         0.0000      1175.0000

instuments CM:    0.0000          6.2500          0.0000
instuments MASS: 1300.0000

ADDING: transferengine INERTIA TENSOR

      13.9900        0.0000          0.0000
          0.0000     10.7800          0.0000
          0.0000         0.0000      13.9900

transferengine CM:    0.0000          -4.4970          0.0000
transferengine MASS: 120.0000

ADDING: retrorocket1 INERTIA TENSOR

      1.2630        0.0000          0.0000
          0.0000     0.3923          0.0000
          0.0000         0.0000      1.2630

[illegible]    [illegible]          -4.[illegible]          0.0000
```

STRC Figure 3. Total Mass and Total Inertia Tensor and Individual Component Mass and Inertias [mass(kg), inertia(kg-m^2)]

```
            1.2630          0.0000          0.0000
             0.0000          0.3920          0.0000
            0.0000          0.0000          1.2630

 retrorocket2 CM:      0.0000        -4.7960        -1.2000

 retrorocket2 MASS:    25.0000

 ADDING: retrorocket3 INERTIA TENSOR

            1.2620          0.0000          0.0000
             0.0000          0.3920          0.0000
            0.0000          0.0000          1.2630

 retrorocket3 CM:     -1.2200        -4.7960         0.0000

 retrorocket3 MASS:    25.0000

 ADDING: retrorocket4 INERTIA TENSOR

            1.2630          0.0000          0.0000
             0.0000          0.3920          0.0000
            0.0000          0.0000          1.2630

 retrorocket4 CM:      0.0000        -4.7960         1.2000

 retrorocket4 MASS:    25.0000

 ADDING: fuelN2O4 INERTIA TENSOR

            80.7300         0.0000          0.0000
             0.0000         80.7300          0.0000
            0.0000          0.0000         80.7300

 fuelN2O4 CM:     0.0000        -2.2660        0.6000

 fuelN2O4 MASS:   788.0000

 ADDING: fuelMMH INERTIA TENSOR

           48.9700          0.0000          0.0000
             0.0000         48.9700          0.0000
            0.0000          0.0000         48.9700

 fuelMMH CM:      0.0000        -2.2660        -0.6000

 fuelMMH MASS:   478.0000

 ADDING: batteries/valves INERTIA TENSOR

            4.1670          0.0000          0.0000
             0.0000          4.1670          0.0000
            0.0000          0.0000          4.1670

 batteries/valves CM:     0.0000        -3.2010         0.0000

 batteries/valves MASS:   50.0000

 ADDING: antenna INERTIA TENSOR

           2.3340          0.0000          0.0000
             0.0000          3.6000          0.0000
           0.0000          0.0000          2.3340

 antenna CM:      3.0790          5.4880          0.0000

 antenna MASS:     9.0000

 Total INERTIA TENSOR

        235828.5790     -10561.7364      -112.3962
          -10561.7364     37480.2255       620.9596
        -112.3962       620.9596     237773.1724

 Total CM:    -0.6042         0.4705         0.0094

 Total MASS: 19859.0000
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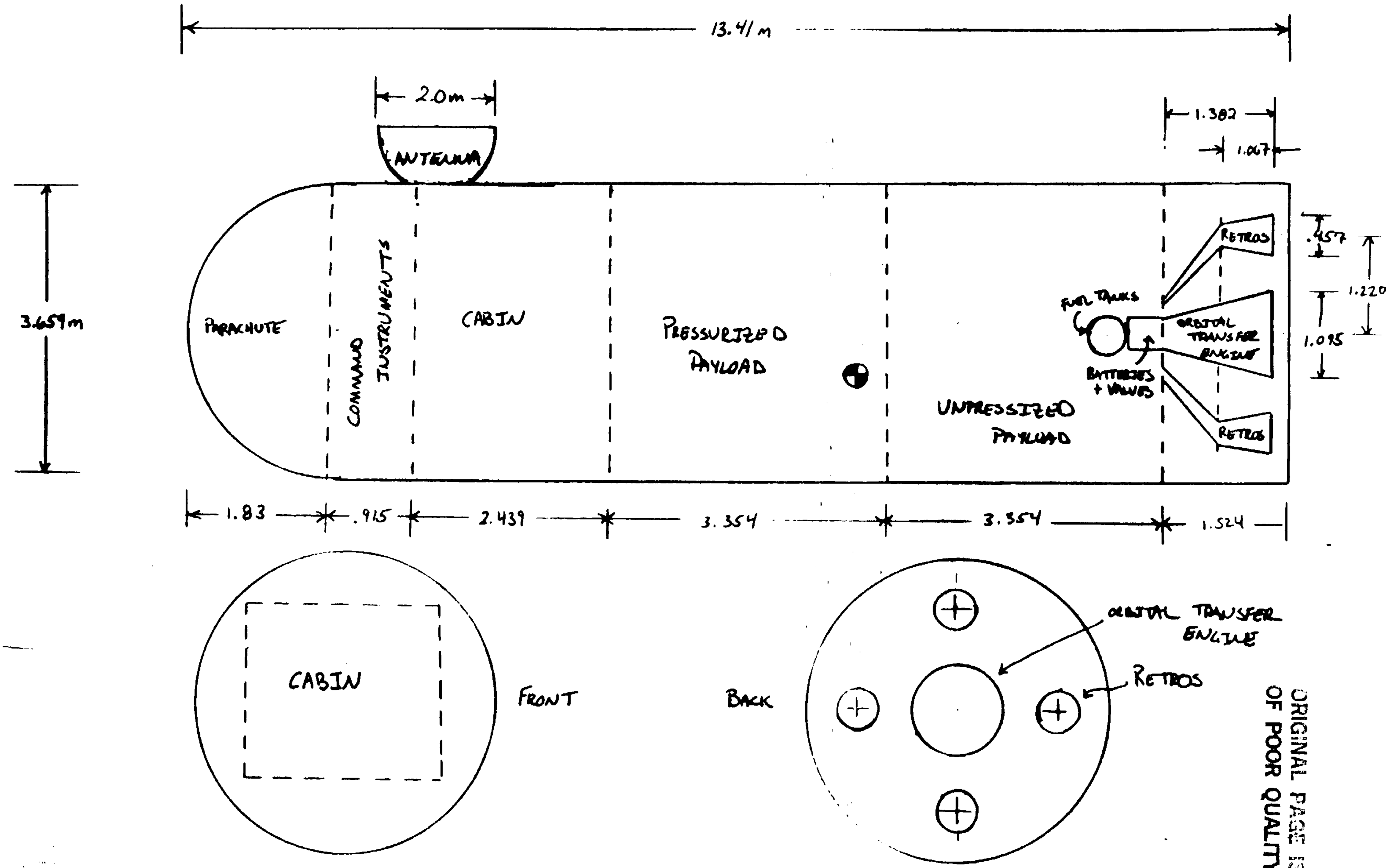

STRC Figure 4. Schematic Diagram of MURPHS. (dimensions in meters)

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1. Shifrin, Carole A., "LTV Begins Designing Composite Wings for AIM-9", Aviation Week & Space Technology, October 6, 1986, pp. 91-93.

2. DeMario, William F., "New World for Aerospace Composites", Aerospace America, October 1985, pp. 36-42.

3. Henninger, John H., "Solar Absorptance and Thermal Emittance of Some Common Spacecraft Thermal-Control Coatings", NASA Reference Publication 1121, April 1984, pp.1-3,8.

PROPULSION

After the TITAN delivers MURPHS into a 185.4 km. orbit, the spacecraft must be boosted to the space station orbit of 290 km. Therefore, an orbital maneuvering system is required to provide the necessary delta-v between these orbits. This delta-v is calculated in Appendix A. The following chart summarizes the potential propulsion subsystem options, giving the positive and negative aspects of each choice.

OPTION	COMMENTS
Nuclear	- Not developed - Social Concerns
Electric	+ High Isp - Low Thrust - Less Developed
Chemical	+ Well Developed, Reliable + High Thrust - Low Isp

The chemical propellant option is the best, because of its development status and high thrust.

Chemical systems fall into two categories: Solid and liquid propellants. Solid propellant systems cannot be throttled or turned off, which is a major drawback to the MURPHS design. A more variable, flexible system is needed for emergency situations. Liquid propellants can be further subdivided into two more groups, monopropellant and bipropellant systems. Monopropellant systems have lower performance characteristics than bipropellant systems. Therefore, from this quick summary, it is obvious that a liquid, chemical, bipropellant engine is the best choice for MURPHS.

A decision must now be made as to which oxidizer and fuel to use. A trade study between the most common oxidizers is in Table PPS 1.

Table PPS 1 : Trade study between oxidizers.

Oxidizer			
	Liquid Oxygen	Fluorine	Nitrogen Tetroxide
Advantages:	High Performance Widely Used Noncorrosive Nontoxic	High specific gravity High Performance	Can be stored indefinitely High Density Used Extensively
Disadvantages:	Very difficult to store Must insulate all materials in contact to prevent evaporation	Very corrosive Very toxic Spontaneously reactive Expensive	Toxic High vapor pressure Narrow liquid temperature range

Nitrogen tetroxide is the only oxidizer which can be stored easily. The only disadvantages of this oxidizer are minor. A slightly heavier tank will be needed to accomodate the higher vapor pressure, and the temperature at which it is stored must be monitored, but these are not major problems. The toxic quality is only a minor drawback if it is kept away from the crew. A trade study between fuels is shown in Table PPS 2.

Table PPS 2: Trade study between liquid fuels.

	Fuels		
	Liquid Hydrogen	Pure Hydrazine	Monomethylhydrazine
Advantages:	High performance Light	Good performance High freezing point	Good thermal properties Good liquid temperature range Most stable hydrazine Proven performance
Disadvantages:	Must be kept cold Low Density Must insulate tanks, lines to prevent evaporation	Toxic Very reactive with many materials	Toxic Reactive

Liquid hydrogen is difficult to handle. Monomethylhydrazine (MMH) is the most stable form of the hydrazines, and has been used extensively.

From the above trade studies, a MMH and nitrogen tetroxide (N2O4) combination is very desirable for the orbital transfer engine. This is also what the proven and very reliable space shuttle orbital transfer engines use. Because of its exceptional reliability and performance, a space shuttle orbital transfer engine, with some significant modifications, will be employed in MURPHS. These modifications will be explained in the next section.

Schematic Description

A schematic of the propulsion system is shown in Figure PPS 1. To pressurize the fuel through the plumbing to the engines, two gaseous helium tanks are used. Tank 1 will be used in most cases and tank 2 is for redundancy. The gas pressure in the tanks is monitored by pressure transducer 1 (PT1) and PT2. The tanks are activated by opening high pressure latching valves (HPLV) numbered according to the tanks. The helium then branches into two paths. The first path, utilizing HPLV3 and pressure regulator 1 (PR1), is normally used, with the option of using HPLV4 and PR2 if a failure occurs in the first path. The helium then splits into two branches which lead to the propellant tanks. Each branch contains a quad-check valve to protect the helium pressurization components from exposure to propellant. A pressure relief valve is also found on each branch in case of pressure overloads. A final valve allows the helium to force propellant out of the tanks. Each tank has its own redundancy option path as used by the helium. In the case of the MMH tank, the fuel will normally pass through HPLV5 and its pressure will be monitored by PT3. The redundant path, with PT5 and HPLV7, are to be used in the event of failure in the primary line. The valves used to allow the propellant into the nozzle are two series redundant ball valves, and are activated by gaseous nitrogen, exactly as in the shuttle system. The high pressure nitrogen tank is also connected to the cold gas reaction control system, for maneuvering near the space station. HPLV9 and HPLV10 control the N2O4 and MMH flow to the retro-rockets, respectively. Valves just above the retro-rocket nozzles control the throttling of these engines. The retro-rockets also rotate on a universal joint for vector thrust and accurate landings.

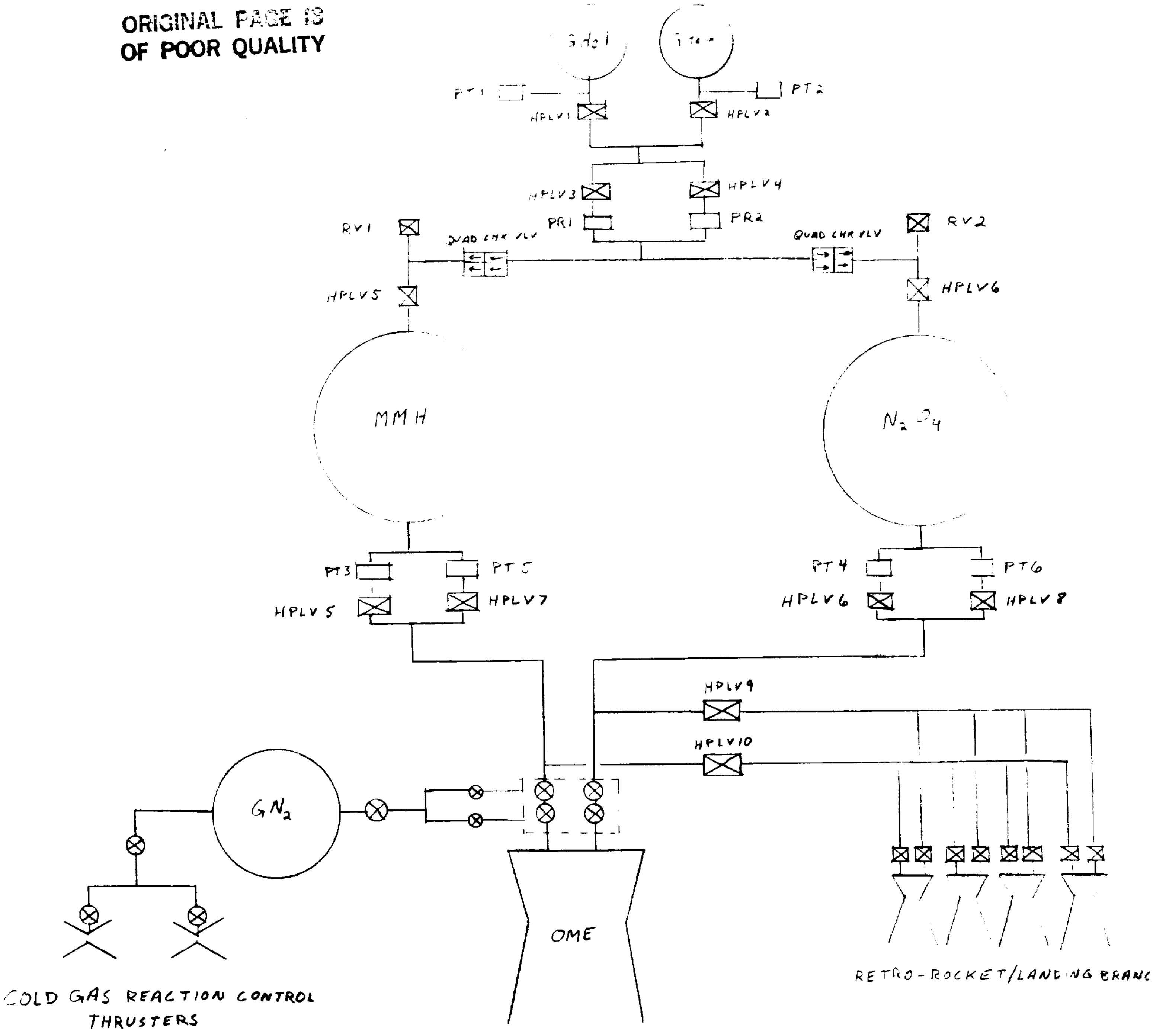

FIG. PPS 1: PROPULSION SYSTEM SCHEMATIC

Interactions With Other Subsystems

AACS: Because no rocket fuel can be burned within the vicinity of the space station, an alternate method of reaction control must be utilized for performing certain maneuvers that the AACS system, using inertia wheels, may not be able to accomplish. Changing speeds is one of these tasks. Cold gas jets a a favorable option because they normally use inert gases which will not corrode the outside of the space station. Nitrogen gives the highest Isp of these gases (80 sec.), and is also a necessary part of the propulsion system because it is needed to operate the fuel valves in the orbital maneuvering engine. Therefore, the size of the nitrogen tank can be increased and reaction control thrusters can be attached.

The main purpose of these jets is to dock and undock. As an example, to back away from the space station, a two second acceleration period from rest to 0.2 ft/sec is needed. This is followed by a 6502 second coast and a deceleration back to rest relative to the space station. The delta-v required for this maneuver is found to be 0.1219 m/sec.

$$\Delta v = g(Isp)\ln(\text{mass initial}/\text{mass final})$$

Mm=Mass of MURPHS; entire mass without fuel and cargo=10826kg
Mu=Mass up; mass of cargo to supply space station=8110kg
Md=Mass down; mass of cargo brought back from space station=6547kg
Mr=Mass retro rockets; mass of fuel used by retro rockets on decent
Mf2=Mass of fuel on second leg of mission; to get back from space station;reentry
Mf1=Mass of fuel used for orbital transfer to get from orbit of 185.4km to the space station orbit of 290km.

$$0.1219=9.8(80)\ln(Mm+Mf2+Mr+Md/\text{final mass})=9.8(80)\ln(10848/\text{final mass})$$

final mass=18045.19kg   mass of nitrogen expelled=2.806kg

Because more maneuvers such as this one will be needed, and some nitrogen will be necessary to open valves, a significantly larger amount of nitrogen can be added without a significant weight penalty. Twenty-five kilograms will be sufficient for MURPHS. This gas will be stored under very high pressure, 3500 psia. The density of nitrogen at this pressure is 17.37 lb/cu. ft. Therefore, the radius of the spherical nitrogen tank can be found.

$25kg(2.205\ lb/kg)(1/17.37)=3.1736\ cu.\ ft.=4PIr^3/3$

$r=.9116\ ft.=10.94\ in.=27.78\ cm.$

Reentry /Recovery:

The reentry delta-v has been calculated to be 100 m/sec. Also, the delta-v necessary for landing, using the retro-rockets, was calculated to be only 10 m/sec. The mass of fuel used on these two parts of the mission is calculated in the next section.

Mission Management:

Because mission management has decided to send two capsules up every ninety days, the mass of cargo up will be 8110 kg, and the mass down will be 6547 kg. Using this information, the following fuel mass calculations can be made. The variables used are the same as those defined in the AACS nitrogen-sizing calculations.

Note: The same fuel system will be used for the retro-rockets as the orbital transfer engine uses to save mass and the complexity of having two different propulsion systems on MURPHS. The orbital transfer Isp of MMH and N2O4 is 310 sec in space. For the retro-rockets, however, because they will have to be fired in the atmosphere, a lower estimate of 250 sec. will be used.

$$\Delta v = g(Isp)\ln(\text{initial mass}/\text{final mass})$$

Δvr=delta-v for retro-rockets=9.8(250)ln(Mm+Mr+Md/Mm+Md)= 10 m/s
0.00409=Mr/Mm+Md=Mr/17373
mass of fuel needed for retro-rockets=Mr= 71.056 kg

$\Delta v2$=delta-v for coming back from the space station
=9.8(310)ln(Mm+Mf2+Mr+Md/Mm+Mr+Md)=100 m/sec
mass of fuel needed to get back from space station= Mf2= 583.75 kg

$\Delta v1$=delta-v needed to get from orbit Of 185.4 km to the space station
=9.8(310)ln(Mm+Mf2+Mr+Mu+Mf1/Mm+Mf2+Mr+Mu)=61.357 m/s
mass of fuel needed to get to space station=Mf1= 399.69 kg

Total Fuel = 71.056+583.75+399.69 = 1054.5 kg
=1265.4 kg with 20% redundancy
Total Propulsion System Mass= 1265.4
\+ 120.0 kg (approximate weight of nozzle)
\+ 20% of these mases (estimate for valves, lines, tanks, etc.)

1662.48 kg = approximate mass of total subsystem

Fuel Tank Sizing:

From the above calculations, the mass and volumes of the MMH and N2O4 tanks can be found.

Total fuel=71.056+583.75+399.69 = 1054.5 kg

An extra 20% fuel will be added for redundancy, bringing the total mass of fuel up to 1265.4 kg. This engine operates at an oxidizer/fuel ratio of 1.65.

1.65 = kg N2O4/kg MMH = y/x   y=1.65x

x+y = 1265.4 kg = 2.65x

x = 477.5 kg MMH   y = 787.9 kg N2O4

Specific gravities:

MMH= 0.8788 kg/liter   N2O4= 1.447 kg/liter

477.5 kg/0.8788 = 543.35 ltr MMH =0.54335 cu. m.

787.9 kg/1.447 = 544.5 ltr N2O4 = 0.5445 cu. m.

$0.54335 = 4PIr^3/3$ $0.5445 = 4PIr^3/3$

r = 0.5062 m   r = 0.50656 m   for spherical tanks

POWER

The possible power system options for MURPHS are solar arrays, batteries, and fuel cells. Fuel cell systems require a thermal energy conversion system. These systems, in general, are heavy, and designed for long-term, continuous operation. The fuel cells used in the shuttle are quite heavy, and produce much more power than is required by MURPHS. Altough a smaller system may be feasible, fuel cells are still not the best choice for MURPHS.

This narrows the decision to batteries and/or solar arrays. Solar arrays can be attached to the actual body of the spacecraft, or they can be deployable on extendable panels. Because the capsule must return through the atmosphere, body mounted arrays are immediately ruled out. Deployable arrays, which can be folded up and protected when they are not needed, are still a reasonable choice. It is not preferable, however, to have the deployable arrays unfolded during orbit transfer maneuvers. During transfer maneuvers, the probability increases of micro-meteorite damage. In addition, thrust impulses could damage the fragile array structure. While the thrusters are burning, the deployed array's natural frequency must be able to withstand the vibrations from the maneuvers. This may require expensive materials. To further emphasize this point, Table PPS 3 illustrates the fact that deployable solar arrays are not specifically designed to deliver power in orbital transfer maneuvers unless it is absolutely necessary. Under normal circumstances, the MURPHS vehicle will be in transfer or reentry orbits for nearly the entire time it is not docked to the space station. This is a definite drawback to the selection of solar arrays for MURPHS. The high cost of materials needed for solar arrays is another disadvantage. In addition, at least a large fraction of the weight savings incurred by the use of solar arrays would be lost on the added complexities required for a deployable system. A drive motor, along with the gear assembly and other related machinery, is needed to unfold the array. An additional protective casing for the folded array must also be fabricated to shelter the array from damage during reentry. Furthermore, for optimum performance, an additional motor, to move the rotating panels, and an attitude control system must be employed to keep the arrays perpendicular to the sun at all times.

These deployable array characteristics are not very compatible with the MURPHS design, which will need power for only short periods of time. If MURPHS takes power from the TITAN during the boost stage, and uses power from the space station while docked, it will need power for only several hours at the most. In addition, by eliminationg arrays, the problems of being shaded by either the earth or the space station, as well as radiation degradation, do not need to be addressed. Now the decision becomes a choice between battery

Table PPS3 Comparison of Different Deployable Solar Arrays.

Order of Merit	Rigid	Semi-Rigid	Flexible Roll-up	Flexible Fold-up (Extendable boom)	Flexible Fold-up (Pantagraph)	Flexible Fold-up (Telescope)
Power to weight Ratio	6	5	4	3	2	1
Stowed Volume	4	4	3	1	1	2
Stiffness	1	1	3	3	2	2
Adaptability	3	3	1	1	1	2
Development Potential	3	2	1	1	1	1
Cost	5	4	3	2	2	1
Power Transfer Mechanism	1	1	2	1	2	2
Power in Transfer Orbit	Some Power	Some Power	No Power	No Power	No Power	No Power
Examples	11 KW Apollo Telescope Mount on Skylab; Symponie	Boeing 46 KW Array	1.5 KW Hughes Solar Array; AEG Solar Array	1 KW Solar Array for CTS	Developed by SNIAS, France	RAE Proto-Type Solar Array

(Deployable Solar Array Types)

Note:— Usually the Solar arrays are deployed in the parking or transfer orbit whenever flexible solar arrays are carried, in the absence of any small body mounted array for generating transfer orbit power.

Taken From <u>Satellite Technology and Its Applications</u>, P.R.K. Chetty

systems. A trade study between battery systems is shown in Table PPS 4.

Table PPS 4: Trade study between batteries.[1]

CRITERIA	Ni-Fe	Ni-Zn	Ni-Cd	Ni-H_2	Ag-Cd	Ag-H_2	Ag-Zn
Specific Energy (Whr/kg)	27	60	30	55	55	80	90
Nominal Voltage per Cell (Volts)	1.2	1.6	1.2	1.4	1.2	1.4	1.5
Temp. Range (C)	-10 to 45	-20 to 60	-20 to 45	0 to 55	-25 to 70	0 to 50	-20 to 60
Cycle Life*	2000-4000	50-200	500-2000	1500-6000	150-600	500-3000	100-150
Energy Density (Whr/Ltr)	55	120	80	60	110	90	180
Approx. Cost $/kWhr		200	400-1000	2000	1000->2000	>2000	800->1500

*Cycle life depends on DOD

NOTE: Lithium systems have been eliminated because their development is not mature enough. They are in the development stage, and have an unproven record in space applications.

If MURPHS uses power from its batteries on the way to the space station, and then recharges them while at the station, and uses the battery power again on the reentry phase of the mission, it will use only approximately 1.5 cycles on every mission. As a result, the cycle life criteria is less important than others. The mass and volume criteria are the most important for the MURPHS spacecraft. This is why the Ag-Zn batteries were chosen. Figure PPS 2 illustrates the difference between the Ag-Zn batteries and several of the other choices. Other benefits of Ag-Zn batteries include a 85% charge retention capacity after 3 months standing at room temperature. This is a useful quality in case MURPHS has to leave the space station quickly, without taking time to get a fresh recharge of the batteries. The Ag-Zn system will still have 85% of its nominal power. In addition, these batteries can be recharged in 10-20 hours.[2]

The only major drawback to these batteries is their cost, and this will be addressed shortly.

Power Estimates:

LSCS	450 W	
CDCS	650 W	
Propulsion	350 W	(Estimate)
AACS	100 W	
	1550 W	= Maximum power of all subsystems

+150 W for redundancy = 1700 W = Maximum power needed for MURPHS

30 hours is the design life for power. This is a very high design life. If MURPHS has to use its own power for this long, something has gone wrong.

$$\frac{1700\text{W}(30\text{ hrs})}{.80(\text{DOD})} = 63750\text{ W-hrs}$$

$$\frac{63750\text{ W-hrs}}{90\text{ Whr/kg}} = 708.33\text{ kg} = \text{Mass of batteries}$$

+20% (wiring, regulators, dc converters, etc.)

= 850.0 kg= Total mass of Power Subsystem

Bus voltage is chosen to be 40 volts   40 volts/1.5 = 26.667 cells

need 27 cells

63750 W-hrs/ 27(1.5) = 1574.07 Amp-hr

$$\frac{63750\text{ W-hr}}{180\text{ W-hr/ltr}} = 354.17\text{ liters of space}$$

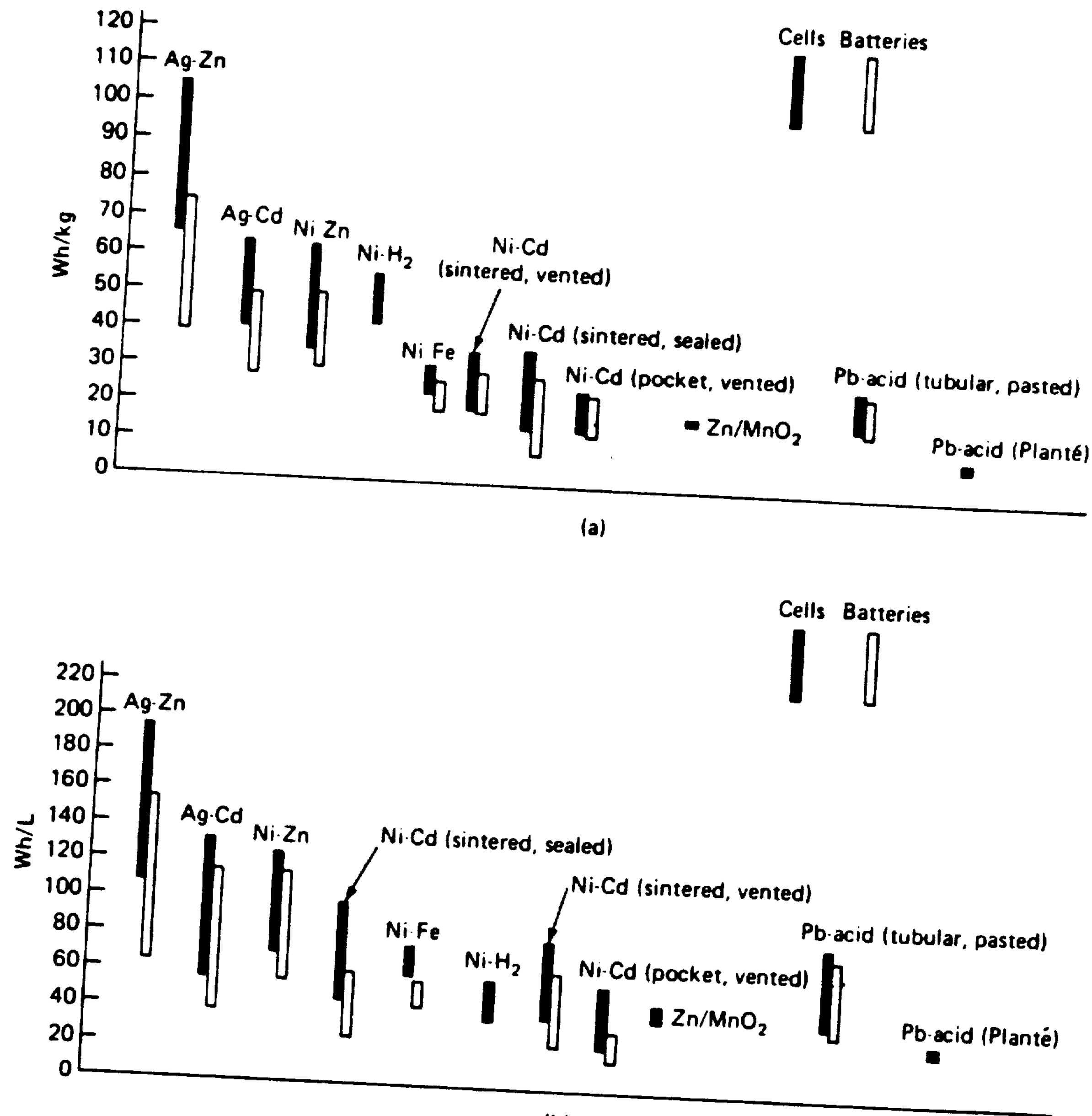

FIG. PP5.2 Energy density of the secondary battery systems at 20°C: (*a*) gravimetric energy density; (*b*) volumetric energy density. (*Adapted from Falk and Salkind.*)

Taken From <u>Handbook of Batteries and Fuel Cells</u>, David Linden

Although the mass is reasonable, and the Ag-Zn batteries do not take up much volume on MURPHS, the cost may still be an issue.
63.750 kW-hrs($1700/kW-hr)= $108375 using a very high estimate of the cost

At this depth of discharge, 80%, a cycle life of 100 can be expected. If the batteries are charged before a mission, discharged on the way to the space station, recharged at the space station, and discharged on the reentry part of the mission, two cycles will be used every mission. This is under normal circumstances. As a result, a battery lifetime of fifty missions is feasible. Because of irregularities, such as emergencies, and a redundancy allowance, a 30 mission lifetime can be expected. This means, because of these worst-scenario lifetime and cost estimates, a maximum of $108375 must be spent on replacing batteries every 30 missions. This is not a very high price considering the increased mass and volume efficiency of this kind of battery.

For power conditioning and control, a Decentralized Regulation Approach (DRA) will be utilized. This means that regulation of power, such as voltage and current regulators, and dc-dc conversions will be carried out at each load end separately. This is the best approach for MURPHS, because the various subsystems, with varying power needs, can individually tailor a system which fits the load requirement or need. A Centralized Regulation Approach that would be compatible with each subsystem would be difficult to design because of the variety and complexities of the different load requirements. Because a DRA will be used, an unregulated main bus will also be employed, with the regulating taking place at each load or subsystem. Furthermore, an unregulated main bus will mean a simpler, lighter power conditioning system.

Failures in load components will be counteracted by parallel redundant fuses on each load. This will prevent danger to the power system. A simple solution to short-circuit failures in the wiring harness is to put double insulation on the system. This is not a guaranteed solution, but it is probably the best that can be devised. If an individual battery cell fails, by open circuit for example, a bypass circuit will skip over that cell. This merely requires fusing individual cells.

REFERENCES

1. Linden, David; Handbook of Batteries and Fuel Cells; McGraw-Hill, 1984, p.13-10-11-15.

2. Crompton, T.R., Small Batteries, Volume 1, Secondary Cells, The MACMILLAN PRESS LTD., 1982, p.163.

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Attitude and Articulation Control

The logistics module and crew emergency vehicle (MURPHS) will be subjected to disturbing torques due to atmospheric drag, solar wind, radiation pressure, magnetic fields, gravity, micrometeorite impacts, components moving in the module, and a spin rate imparted by the Titan IV. MURPHS must maintain a desired attitude and orbit position to be able to rendevous with the space station, point equipment in the proper direction, and to prevent catastrophic tumbling. Attitude and articulation control thus comes into play to control the spacecraft's attitude, control the pointing devices, and also to load and unload the payload, and align the module for docking.

A typical attitude control system is shown in AACS Figure 1. The attitude of the spacecraft must be measured using various sensors on board MURPHS and then corrected by using actuators. A discussion of the different classifications of attitude control systems, sensors, and pointing devices and the choices for MURPHS follows.

A stabilization system must be chosen. The degree of completeness of attitude control, the controlling moments for angular motion, and the method for obtaining signals must be decided to best suit our design. An autonomous system shall be used to fulfill initial requirements for the design. A three-axis system is needed so antennas and other instruments

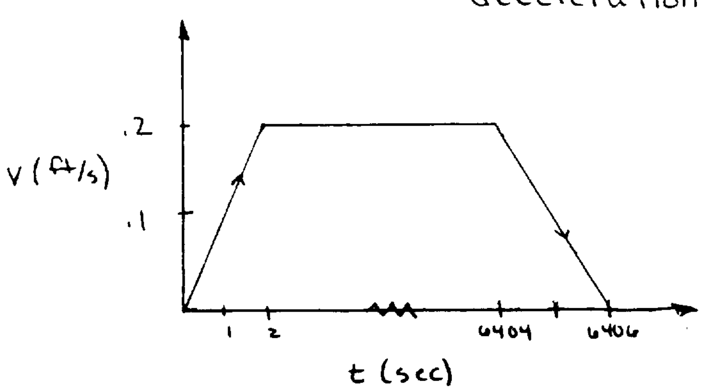
can be pointed in more than one direction at a time.[1] This will allow more fine pointing control.

There are different types of three-axis systems that now must be decided upon. A choice must be made between an active, passive, or a combined system (AACS Figure 2). The decision is based on accuracy, control, response time, operating conditions, cost, and life expectancy. The advantages and disadvantages of all three systems can be seen in AACS Table 1. MURPHS would best benefit from a system that could allow a fast response near the space station, but could rely on a slower system once it maneuvers away. Simplicity and a lifetime of at least six years will be stressed. The use of consumables must be kept within limit and must be safe to use near the space station. The weight and power usage of the system must fit other subsystems' requirements. The system must be accurate but must fit into a budget. Keeping all this in mind, the system employed for MURPHS will be a combined system. A three-axis magnetic attitude control system will be used during flight and nitrogen thrusters will be used close to the space station.

The three-axis magnetic attitude control system is appropriate for low orbit spacecraft for which the 290 kilometer altitude of the space station fits. It has a pointing accuracy of better than .5 degrees in all axes.[2] It requires no expendables, and has an acquisition capability that is practically independent of any initial conditions on any or all axes. The use of a magnetic system insures its

economic value and a long service life. The system includes a scanwheel, magnetometer, three magnetic torquers, and control electronics.[3] The weight and power for each is given in AACS Table 2. This system is used for the despin mode and the on-orbit mode. Near the space station a faster response is needed, thus nitrogen thrusters are employed. These thrusters will act under positive space station control. This thruster aided portion is used when approaching and leaving the space station at distances less than 500 meters as seen from the following maneuver:

To back away from space station maneuver involves : 2 sec. acceleration from rest to .2 ft/s
6502 sec. coast period
deceleration back to rest



Distance equals area under curve.

$$d = 2\left(\tfrac{1}{2}\right)(2\,s)(.2\,ft/s) + (6502\,s)(.2\,ft/s)$$
$$= 1300.8\,ft$$
$$= 396.48\,m$$

Distance away from space station when back at rest relative to it is 396.48 m.[4]

Sensors must be used to measure the attitude of the spacecraft. These fall into two types: celestial and inertial. Assorted types of Earth sensors, sun sensors, and star sensors fall into the celestial category. Magnetometers, gyroscopes, and accelerometers fall into the inertial category.

The scanwheel discussed earlier is used for horizon

sensing. However for greater accuracy and since we are using a combined active and passive system, additional celstial sensors will be used. A Boresight Limb Sensor, a type of earth sensor, will be mounted on the payload. This will maximize scientific return as it measures both the displacement angle and the rotational angle as illustrated in AACS Figure 3. It provides real time position information at a rate of four updates per second.[5] It has an absolute accuracy of .09 degrees at the space station's 290 kilometer altitude. The size, weight, and power requirements are shown in AACS Table 3. A star tracker will also be used. The system selected is the Ball Aerospace Standard Star Tracker. It is a fixed head star tracker that is a standard component for NASA.[6] There are no constraints on the spacecraft orientation and it is very versatile as well as sensitive. Some specifications for this fixed head star tracker are shown in AACS Table 4.

The attitude of the spacecraft must be correctable once sensed. The dynamics of the angular motion of the spacecraft around one of its axes is described by the equation:

$$I(dw/dt) = M + M_i$$

I is the moment of inertia of the spacecraft

w is its angular velocity

M is the moment of external forces

M_i is the moment of internal forces of the moment of dynamic reaction of internal rotating masses.[7]

Control systems are designed with the stipulation that the controlling moments, developed by actuators, exceed the perturbing moments which act constantly on the vehicle.[8] External and internal force moments are used as the controlling moments. The actutator angles and rates must be sensed and controlled using gyros, accelerometers, etc. These rotation angles and rates are used in calculating torque and in the control laws, using the following equation:

$$T(t) = J\ddot{\theta} + B\dot{\theta} + K\theta$$

θ is the rotation angle

$\dot{\theta}$ is the rotation rate

$\ddot{\theta}$ is the rotation acceleration

J is the inertia

B is ~~the~~ proportional to the damping ratio

K is the spring constant

T is the torque.

Gyroscopes will be used to provide stability, to provide precession, and for the gyroscopic moment. Rate gyros will be used to measure the spacecraft angular rate and an integrating rate gyro will be used to measure the spacecraft angular displacement. These gyros will eliminate the rotation around the longitudinal axis of MURPHS. The maximum accuracy which may be achieved is estimated at a drift of .1 degrees/hour.[9] The power consumption is only about thirty-one volts dc.

The magnetic torquers described in the three-axis system earlier are a type of actuator. For additional control we

will also use a magnetic bearing reaction wheel. Due to the low volume, low weight, low cost, and high reliability as shown in AACS Table 5, the extra control will not encumber any other subsystems requirements. The system uses a rotating wheel suspended by means of magnetic forces (AACS Figure 4). This wheel can accelerate to achieve a moment of inertia in one or the other direction, thus rotating the spacecraft accordingly in the opposite direction. It need only be mounted on one side in the equipment portion of MURPHS.[10] This additional actuator will help increase the ability to correct the attitude and see that it stays in the desired attitude.

Employing a fleet of five modules, two at the space station, two on the ground, and one for testing, requires that the payload up and down mass be:

payload up mass = 16220.92 kg - 2 vehicles = 8110.46 kg.

payload down mass = 13094.30 kg - 2 vehicles = 6547.15 kg.

The payload must be loaded and unloaded through a hatch with dimensions of 1.27 meters x 1.27 meters. The loading and unloading will occur at the space station and on Earth. Due to the effect of zero gravity many methods such as conveyor belts become infeasable. Romote control arms on rotating bases cannot unload all the payload through the hatch. At the space station the best method of unloading and loading would be to manually carry the supplies and equipment. This should be a relatively easy task in the zero gravity

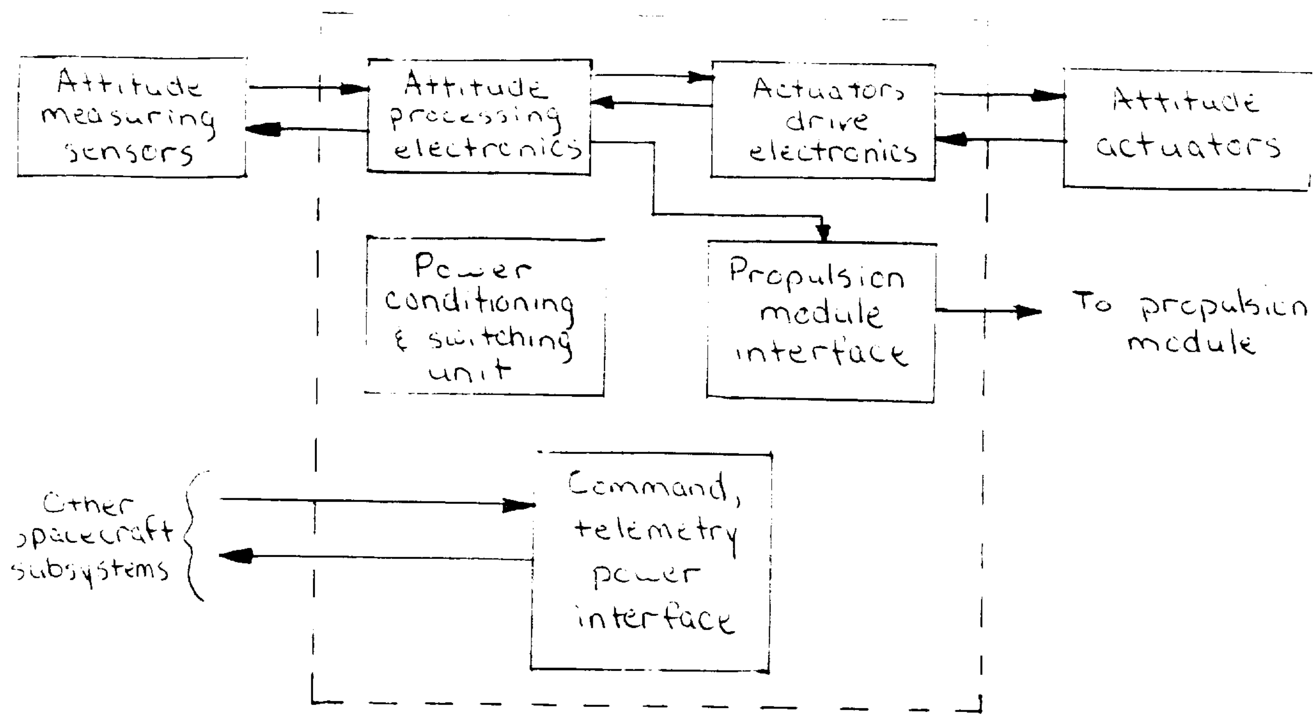
atmosphere. Due to the design of MURPHS the pressurized payload must be unloaded first, then the unpressurized. Loading will be just the opposite of this. The hatch is placed right at the pressurized payload area making the task even easier.

The systems employed by attitude and articulation control affect many other subsystems and are affected by their requirements also. Mission planning determined the number of vehicles to be used, thus determining the payload mass requirements for each module. This was taken into consideration for loading and unloading methods. Structures needed the placement of the equipment (mainly in the payload and equipment areas) and the relative size and weight. The total weight of the attitude control systems is less than thirty-five kilograms. This is very small in comparison to other subsystems. This weight was taken into consideration and incorporated in with the other instruments in calculating the centroid and MURPHS' entire mass for launch purposes. The total power requirement of the attitude control systems is 97.5 Watts and sixty-six Volts dc. Two nitrogen thrusters are also included. The nitrogen tank (I_{sp}= 80 seconds and at our pressure ρ = 260.55 kg/m) required is a twenty-five kilogram spherical tank with radius of .278 meters. Reentry also needs to insure stabilization of the module to prevent catastrophic tumbling and control the spinning, thus making attitude control very important. Command and data control is in charge of docking, but needs to insure a quick response

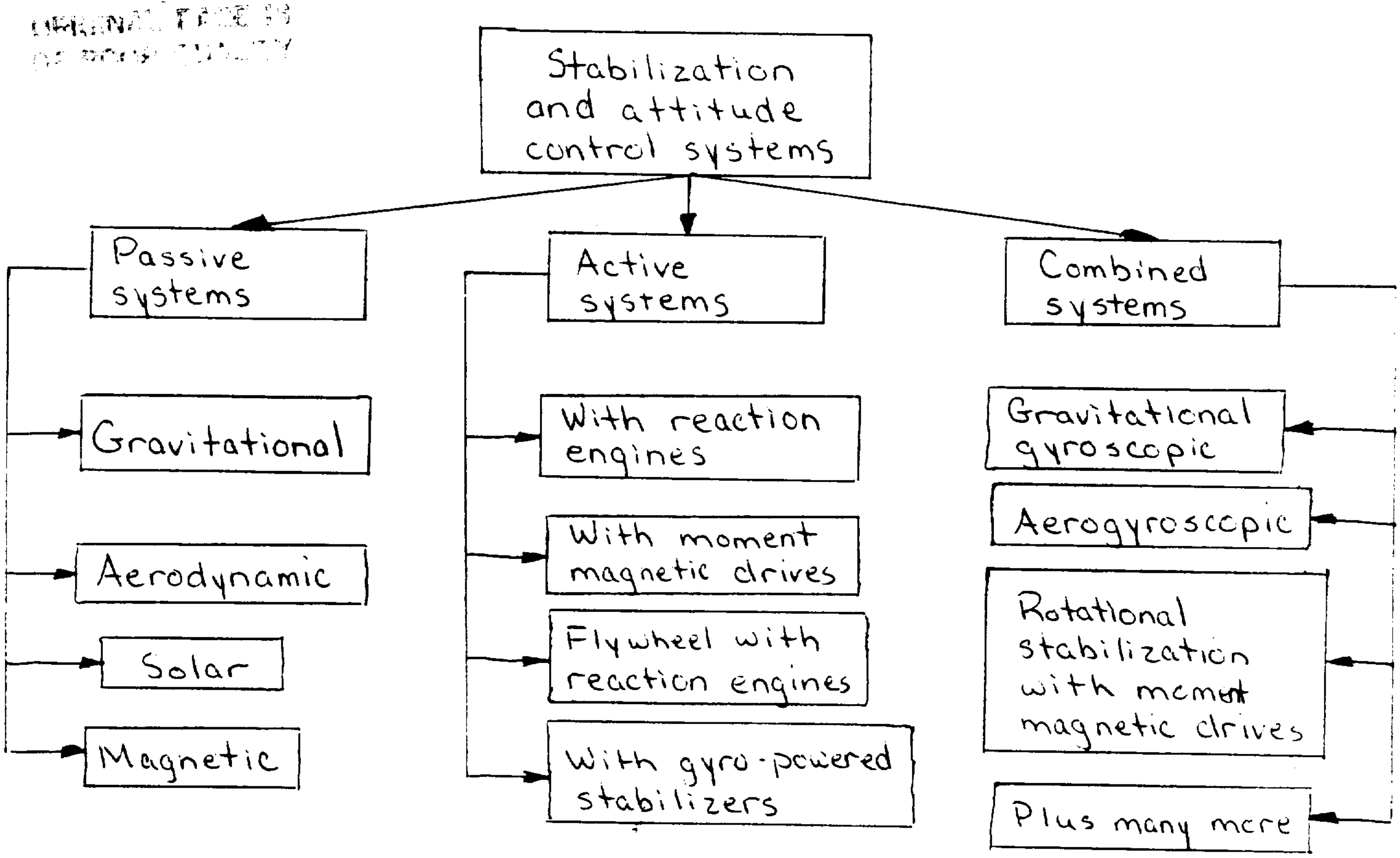
for alignment changes, thus requiring the semi-active system. Overall the attitude control system meets all the other systems requirements as well as fulfilling its own duties.

Footnotes

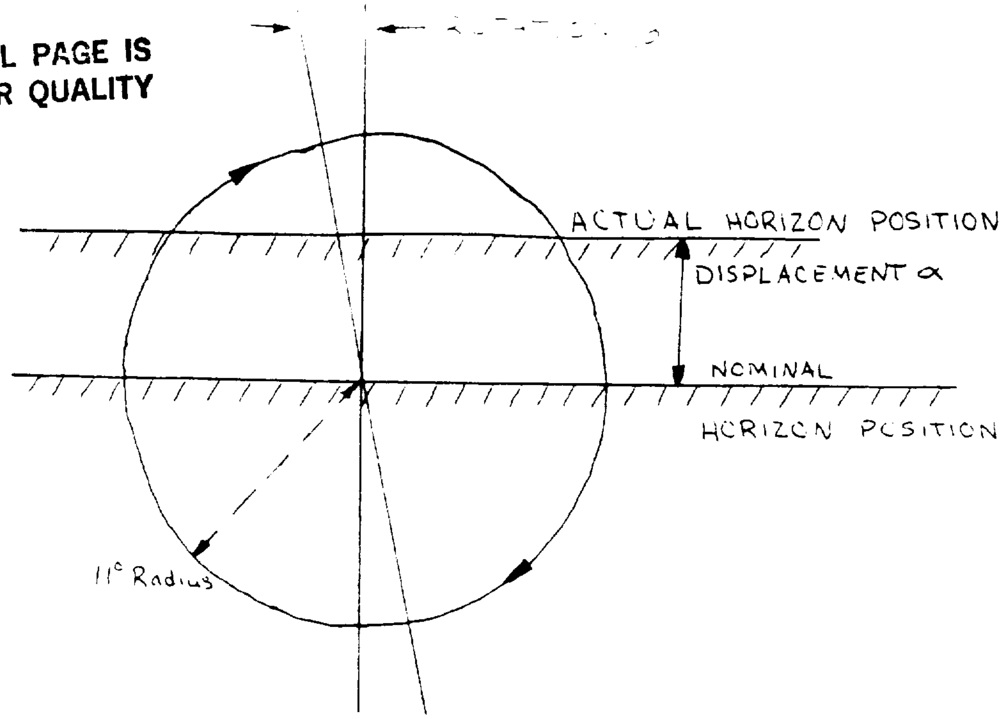
1. Chetty, P. R. K., Satellite Technology and its Applications, p. 156.
2. Ibid., p. 198.
3. Ibid., p. 198.
4. Lembeck, Mike, AAE 241, Spring 1989, Homework Six.
5. Chetty, p. 167.
6. Ibid., p. 172.
7. Korovkin, A. S., Sppacecraft Control Systems, p. 19.
8. Ibid., p. 20.
9. Ibid., p. 52.
10. Chetty, p. 189.
11. Ibid., p. 160.
12. Korovkin, p. 12.
13. Chetty, p. 168.
14. Ibid., p. 188.
15. Ibid., p. 199.
16. Ibid., p. 169.
17. Ibid., p. 173.
18. Ibid., p. 191.



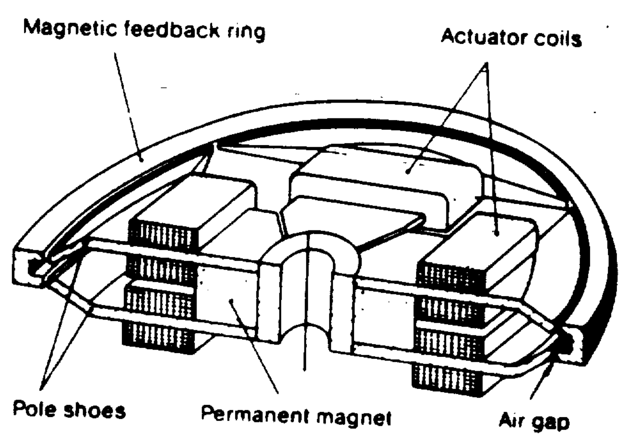
AACS Fig. 1. Block Schematic of Attitude Control System."



AACS Fig. 2. Classification of 3-axis stabilization systems[12]



AACS Fig 3. Boresight limb sensor rotating scan illustration.[13]



AACS Fig 4. Magnetic bearing reaction wheel (isometric).[14]

AACS Table 1. Merits & Demerits of Various Stabilization Methods

Stabilization Method	Merits	Demerits
Active	Great accuracy, fast response, flexible, independent of shape and location.	Costly, limited life, uses great amount of consumables, complex.
Passive (PMB)	Safe operating conditions, unlimited life, allows fine control, cost efficient.	Slow response, nutation, lower accuracy.
Combined	Long life, consumables use limited, accurate, economic.	Slow at times, normally must be used near Earth.

	WEIGHT	POWER
SCANWHEEL	6.8 Kg	2.6 W
ELECTRONICS	3.4 Kg	5.3 W
3 TORQRODS	.35 Kg ea.	.7 W ea.
MAGNETOMETER	.5 Kg	.7 W
TOTAL	11.75 Kg	10.7 W

AACS Table 3. Boresight Limb Sensor Specifications [16]

ATTITUDE RANGE	100 Km and higher		
ABSOLUTE ACCURACY	.1° (250 Km alt) .09° (290 Km alt)		
	SENSOR	ELECTRONICS	TOTAL
SIZE	10.2 cm × 7.6 cm dia	15.2 cm × 17.8 cm × 4.3 cm	
WEIGHT	.93 Kg	1.16 Kg	2.09 Kg
POWER	4.4 W	3.2 W	7.6 W

AACS Table 4. NASA Standard Fixed Head Star Tracker. [17]

FIELD OF VIEW	8° × 8°
SIZE	16.8 cm × 18 cm × 31 cm
WEIGHT	7.71 Kg
POWER INPUT VOLTAGE	21 - 35 Vdc
CONSUMPTION	18 W

AACS Table 5. Magnetic Bearing Reaction Wheel Data

Angular momentum	max ± 2 Nms
Bearings radial	electromagnetic
axial	permanent-magnetic
tilting	permanent-magnetic
emergency	sliding
Slew Rate	≤ 5 deg/sec
Power steady state	3 W
max. torque	< 50 W
Dimensions diameter	< 22 cm
height	< 9 cm
Total weight	3 Kg
Reliability (10 years)	> .96

The **Environmental Control and Life Support System** (ECLSS) is responsible for sustaining life in the CERV in case of partial and/or total evacuation of the crew of the Space Station. Some of the possible threats which would require leaving the Space Station include fire, contamination, injury/illness, an explosion or depressurization. In addition, it should be capable of pressurized cargo transport. The design centers around providing a "shirt-sleeve" environment, i.e. air composition of 21% Oxygen and 79% Nitrogen at 70^{0} Fahrenheit and 14.7 psi. The [redacted] seven **subsystems** to be considered are as follows:

Temperature and Humidity Control (THC)
Atmosphere Control Supply (ACS)
Atmosphere Revitalization (AR)
Water Management (WRM)
Waste Management (WM)
Fire Detection and Suppression (FDS) and Medical Support.

Our mission is designed to accomodate eight people (8) for **thirty hours (30)**. Based on the fact that **two (2) CERV** will be at the Space Station at all times, the information that follows is for a **four (4) man vehicle.** The cabin size will be 12.4 cubic meters - **2.4 x 2.4 x 2.15 m *.**

The relative simplicity of the design is based on the fact that 1) mission length is short, <= 30 hours, and 2) simplicity provides greater reliability, shorter turnaround time and use of "tried and true" technology. Perhaps the best reason is that this vehicle is to be **used in emergencies only.**

Consumables to Accomodate and Their Quantities

	Equations for calculations
LiOH Usage - 6.9 kg	3.0(lb/man-day)(x men)(y days)(1 kg/2.2lb)= kg
N_2 Leakage - 4.5g	0.33(lb/hr)(z hours)(1 kg/2.2 lb) = kg
O_2 Usage - 4.8 kg	2.08(lb/man-day)(x men)(y days)(1 kg/2.2lb)= kg
Water Exhaled - 12.6 kg	5.5(lb/man-day)(x men)(y days)(1 kg/2.2lb) = kg
Metabolic Heat - 67.5 kJ	533(BTU/man-hr)(x men)(z hours)(1054kJ/BTU)= kJ
Food - 10.3 kg	4.5(lbs/man-day)(x men)(y days)(1 kg/2.2 kg)= kg

* Cabin Sizing Calculations

V min: $[-(.004)y^2+1.4219y+81.307](x\ men)(y\ days)(.3048\ m/ft)^3=11.8\ m^3$

V max: $[-(.0068)y^2+2.8346y+83.44](x\ men)(y\ days)(.3048\ m/ft)^3=12.4\ m^3$

Choose **Cabin Volume = V max = 12.4 m^3**

For our calculations, x = 4, y = 30/24 = 1.25, z = 30

ORIGINAL PAGE IS
OF POOR QUALITY

TEMPERATURE AND HUMIDITY CONTROL

* 2 16" oscillating standing fans circulate and cool air within cabin
* 1 wall thermometer monitors cabin temperature for manual control of fan speed
* after CO2 removal, thermostat checks air temperature and automatically controls speed of fan which is cooling air in circulation system
* 25 pint dehumidifier with automatic humidstat removes moisture from air before it is vented into the cabin
* food will be limited to "add water only" dry goods
* any equipment cooling that needs to be done will be vented to the atmosphere through cold plating

Sizing of dehumidifier :

H_2O exhaled = 12.6 kg

Converting units, we obtain
12.6 kg = 12600 g = 12600 cm3 = 12.6 L = 12.6 L(1 Gal/ 3.786 L) = 3.4 Gal
3.4 Gal = 217.6 ounces = **13.6 pints = H_2O exhaled**

13.6 << 25 pints capacity of Samsung's dehumidier fi

Cost Estimation

*	$150	3 fans	20 kg
*	$130	1 Samsung Dehumidifier	21 kg
*	$ 50	Food	10.3 kg
*	$ 25	1 Cabin thermometer	2 kg
*	$100	1 internal thermostat	5 kg

TOTAL COST $455
TOTAL WEIGHT 60 kg

ATMOSPHERE CONTROL AND SUPPLY

We will make use of pressure vessels instead of cyrogenic vessels due in large part to their extensive shelf life. They will be made of stainless steel, density of 0.28 lb/in ^{3} and S_y = 30 ksi. We will be working with a safety factor of 3. These tanks will be stored at 80 ^{0}F under 3000 psi.

Nitrogen Tank Sizing ** R=54.15 (ft^3-lb$_f$/lb$_m$-^{0}R)

r_i = 7.214 in ≅ .19 m

t = 2.55 in ≅ .06 m

r_o = 9.8 in ≅ .25 m

volume = 2510 in^3 ≅ .05 m^3

mass = 705 lb ≅ 321 kg

stress: $\sigma_{t,max}$ = 10,000 psi

$\sigma_{t,AV}$ = 8,500 psi

$\sigma_{A,AV}$ = 3,610 psi

Oxygen Tank Sizing ** R=48.28 (ft^3-lb$_f$/lb$_m$-^{0}R)

r_i = 8.84 in ≅ .23 m

t = 3.12 in ≅ .08 m

r_o = 11.96 in ≅ .31 m

volume = 4607 in^3 ≅ .08 m^3

mass = 1300 lb ≅ 591 kg

stress: $\sigma_{t,max}$ = 10,000 psi

$\sigma_{t,AV}$ = 8,500 psi

$\sigma_{A,AV}$ = 3,600 psi

**** Computations** **USE PV=(Mass)R T**

V=Mass(RT)/P=x lb$_m$(R)(T+460^0)/(P(144in^2/ft^2)= in^3

V=Pi(r_i^3) ----------→ r_i= in

t = (Pr_i/10000) - (P/2) = in

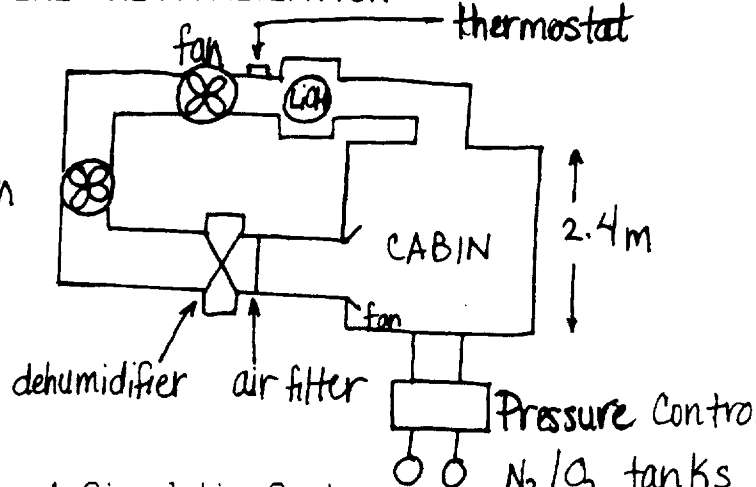
$S_y = \sigma_{max}$(FS), = Pr_i/t psi, =P(r_i)^2/t(t+2r_i)psi

Vol tank = $\pi(r_o^2-r_i^2)r_i+2\pi r_o^2 t$, Mass tank = ρ(Vol) lb$_m$, 1 in = .0254 m

TOTAL WEIGHT 920 kg

ATMOSPHERE REVITALIZATION

Recirculation system will lie above the cabin.



LSCS Figure 1: Circulation System

We will use **6.9 kg of LiOH** for removal of CO2 from the air as shown by

$$2LiOH + CO2 \text{-----} LiCO3 + H20\ (g) + Q$$

This process will occur immediately after the air is drawn into the recycling system. Air filter will use a 6 stage prefilter to purify the air before it is vented back into the cabin.

TOTAL COST	$75
TOTAL WEIGHT	10 kg

WATER RECOVERY AND MANAGEMENT

Due to mission length, water recovery is not a concern but we will need to accomodate potable water for the crew.

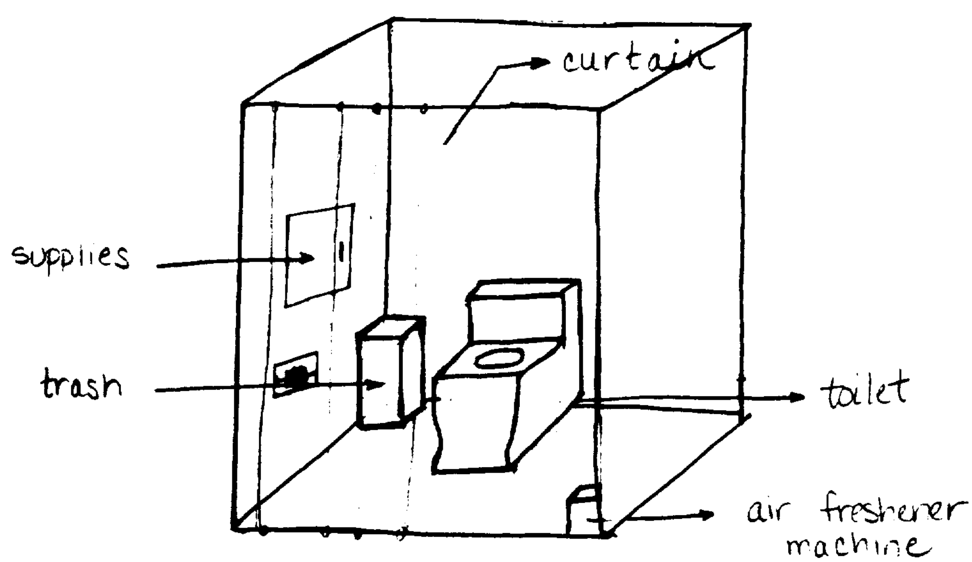
A person should drink 1 gallon of water per day. But for our purposes we will restrict this value to 0.5 gal per day.

Total potable water needed (To be stored in plastic bottles)
0.5 gal (3.786 L/gal) 4 (people) = 7.6 L = 7.6 kg

TOTAL COST	$2.00
TOTAL WEIGHT	10 kg

Waste Management

For this aspect of ECLSS, the design calls for a porta potty environment to be designed for the cabin. Human waste will simply be stored in individual plastic bags which will be thrown into a trash receptacle. The plastic bags will be placed in the stool's base to be inserted at each use. Handwashing will occur at the hands of individually packed travel handy wipes to be easily thrown in the trash. The trash cans will differ from those found on earth only in that they will be securely fastened and have intermediate flaps along the inside to prevent losses due to the zero gravity environ. The top will have a secure closure. A container of this nature will also exist in the main cabin. To prevent odors, a clean air machine will be installed here as well.



LSCS Figure 2: Porta Potty Design (.6x.6x2.15 m)

TOTAL COST $200

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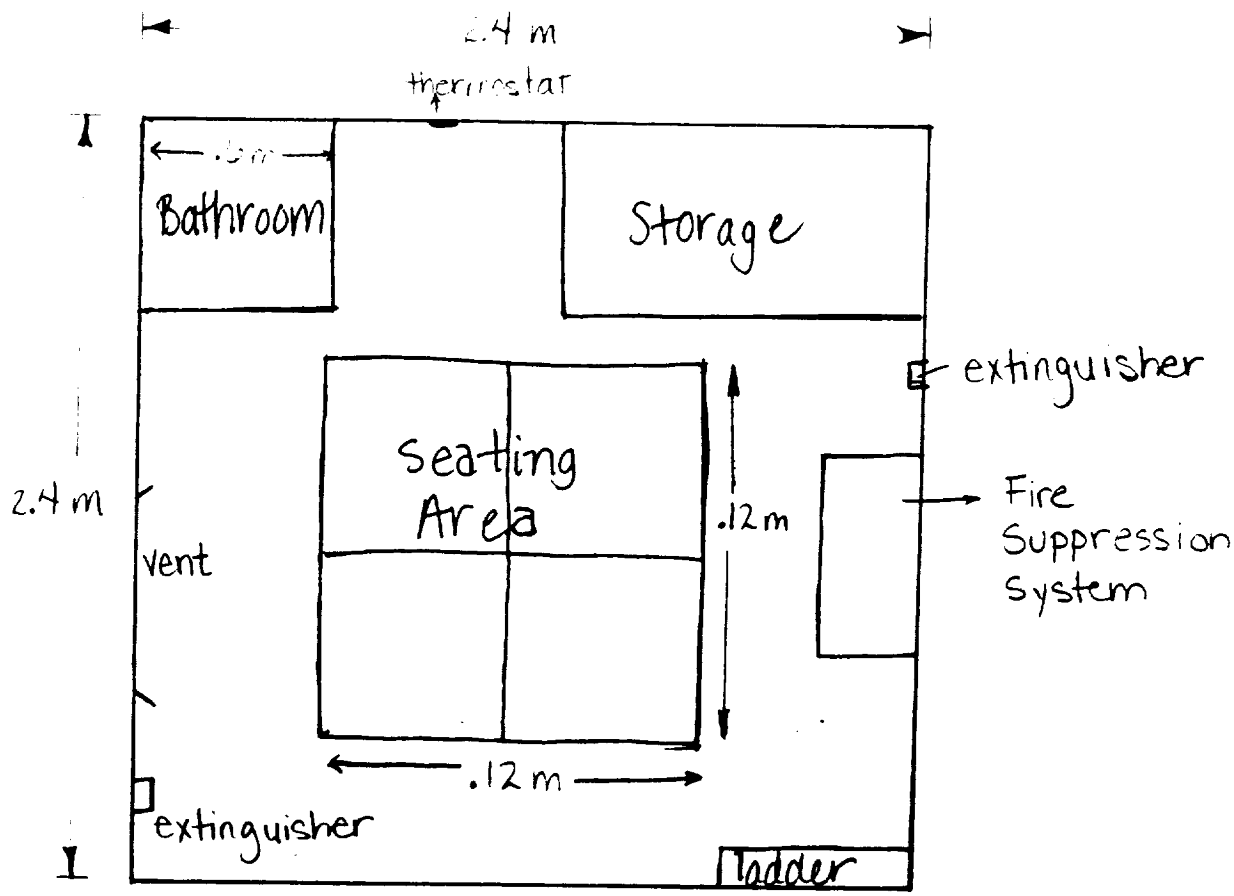
LSCS Figure 3: CABIN LAYOUT



FIRE SUPPRESSION/DETECTION

It is the recommendation of this group to install an AUTOPULSE2000 integrated fire detection/suppression system made by ANSUL. This system will make use of **both an ionization detector and a photoelectric detector** in order to cover most every type of incident that will cause a fire. 1 of each will be necessary as 1 detector is needed for every 7.1 m^3 (12.4/7.1 = 2 detectors). **Halon is the primary extinguishing agent** and is best known for its wide use where electronic equipment is present. 10 kg of Halon will be needed because .5 kg of Halon are needed for every 1.5 m^3 (12.4/1.5 = 8.3 kg Halon). In addition, 1 small hand held extinguisher will be stationed on each side of the cabin for a total of 2. Other benefits of Halon are that it is colorless, odorless, fast-acting, clean and safe for humans.

Following are some of the reasons why other systems were **not** chosen. Water sprinklers would provide extensive damage to scientific equipment. Foam as an extinguisher is difficult to clean. Chemical extinguishing agents may cause equipment corrosion and an irritating cloud of dust for humans. A carbon dioxide based system is no good because it would eliminate O_2 from the protected area. Ultrasonic wave detectors are too easily affected by rapid air movements in a small area. Heat radiation detectors need a direct line of vision.

Cost Estimate **$2000**
(Based on estimate of $5000 for a system of 100 lbs and 10 detectors)
Total Weight **60 kg**

References Langdon-Thomas, G.J. Fire Safety in Buildings - Principles and Practice. St. Martin's Press, New York, c.1973. pages 152-178.
Information provided by Ansul Fire Protection - *Halon Fire Suppression Systems* & *Facts About Protecting Electronic Equipment Against Fire*.

MEDICAL SUPPORT

Medical support will be kept minimal because of the short duration of flight and Space Station support is available should the need arise. Contents of the cabinet will be similar to that found at home including

aspirin
ointment (Bacitricin)
bandaids/gauze
splint/sling
antacid
and rubbing alcohol.

TOTAL COST $25

INTERACTIONS WITH OTHER SUBSYSTEMS

STRC- discuss size of cabin, placement and shape
MMPC - mass, volume, approx. cost
PPS - power requirments
AACS - how to remove crew when docked (ladder leading from cabin floor to access the supplies area and hatch to unload)

FOR PPS:

THC & ATC	fans 3x70 W		210W
	dehumidifier		50 W
	Temp. Controller		20 W
	Air filter	50 W	
FSD	detectors		50 W
WM	aircleaner		70 W
	Total		450 W

FOR MMPC

Total Volume	12.4 m
Total Weight	1100kg
Total Cost	$3500

Bibliography

1. Chetty, P. R. K.; Satellite Technology and its Applications; TAB BOOKS, Inc; 1988.

2. Kaplan, Marshall H.; Modern Spacecraft Dynamics and Control; John Wiley and Sons Inc.; 1976.

3. Korovkin, A. S.; Spacecraft Control Systems; NASA Technical Translation; Report F-774; May 1973.

REENTRY AND RECOVERY SYSTEMS

The Logistics Resupply and Crew Emergency Return Vehicle must be designed for a safe reentry and recovery with the possibility of a high mission frequency over a large period of time. The reentry and recovery system of the vehicle must protect the crew and cargo from the high thermal loads and the considerable g-forces encountered during a ballistic reentry and provide a reasonable time to recovery. The M.U.R.P.H.S. concept of a ballistic reentry and a parachute-retro-rocket land recovery is designed to meet such requirements.

The M.U.R.P.H.S. module will deorbit from Freedom's altitude with a delta-V of .10 kg/sec. The reentry trajectory options are shown graphically in figure RRS 0. The entry flight path angle, γ , becomes the trajectory design driver because of its inverse relation with both delta-V and maximum acceleration. An entry flight path angle of 1.25 degrees approaches the low end of safe entry and yields a maximum acceleration of 3.5 g's, an entry velocity of 7.875 kg/sec, and a delta-V of .10 kg/sec. This trajectory will put the module on its way to a ballistic reentry.

The M.U.R.P.H.S. module will reenter the Earth's atmosphere, at an altitude of 121.9 kg, by the proven method of ballistic reentry. The M.U.R.P.H.S. concept of the reentry vehicle calls for a bullet-shaped module. The module will enter the Earth's atmosphere in a sideways manor with

Flight Path Angle vs. g-Force and ΔV

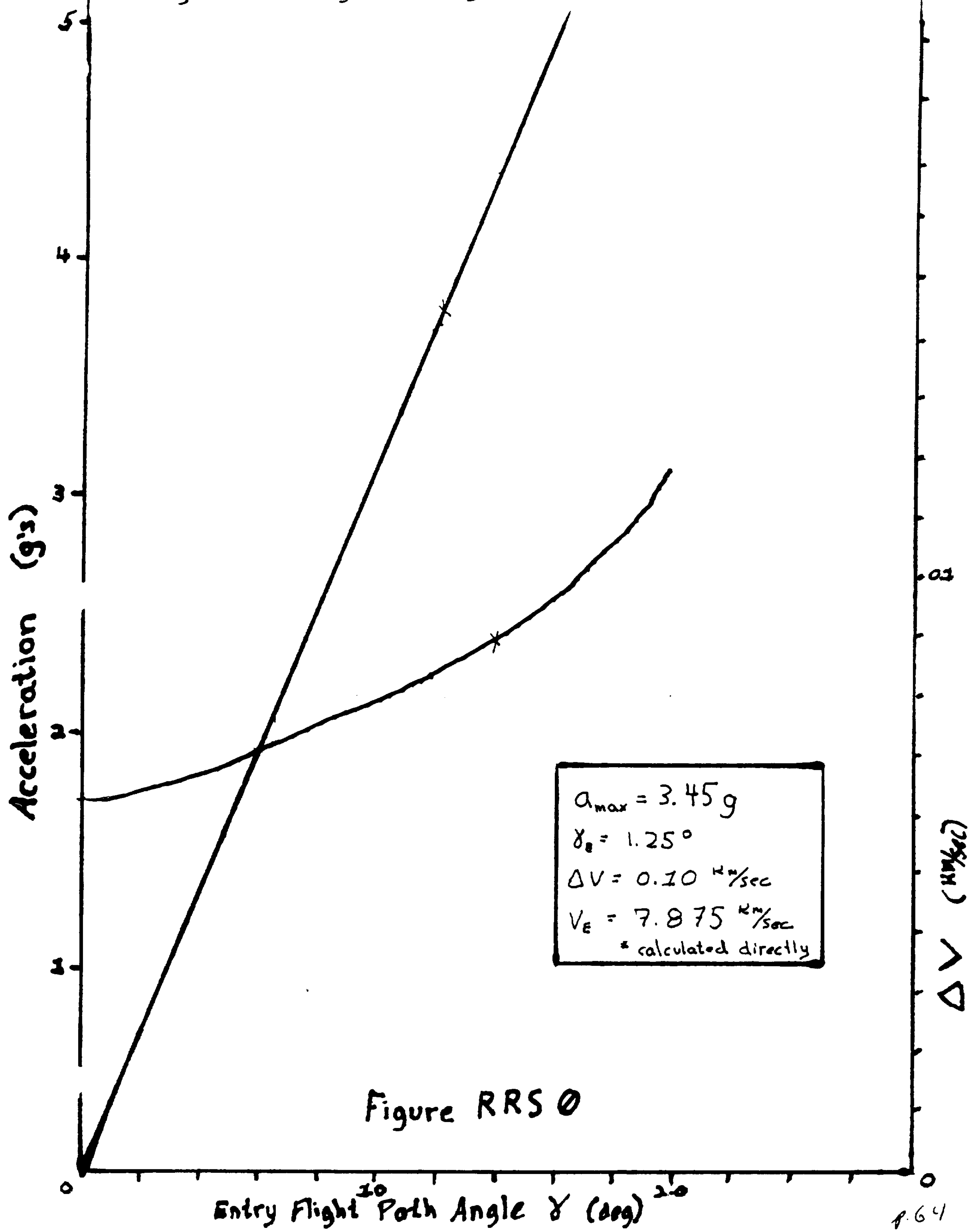


Figure RRS 0

the spherical cap referenced as the top. The center of mass of the vehicle is placed so as to cause the proper surface of the vehicle to absorb the momentum during reentry. The center of mass placement will bring the vehicle to a stable equilibrium at an angle of attack of 60 degrees. This is depicted in figure RRS 1. The induced moment in both directions around the designed center of mass will be equal when the angle of attack is about 60 degrees. At this angle of attack the coefficient of drag on the module will be about 0.71. A complete analysis of drag on inclined cylinders is provided in Hoerner[1].

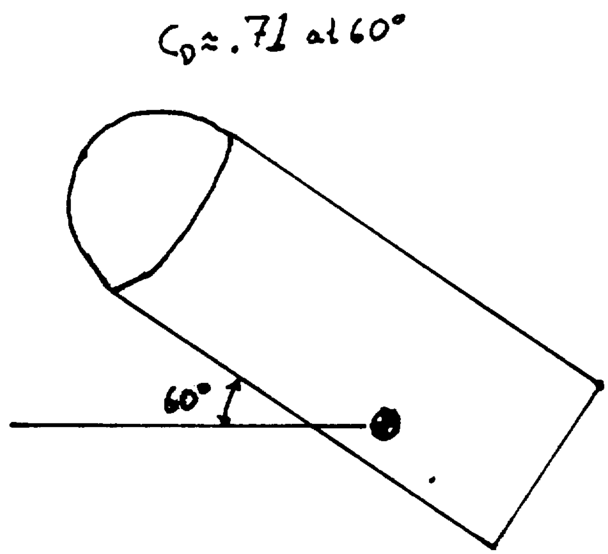


Figure RRS1. Reentry Orientation

The heat shield of the reentry vehicle will be constructed of reinforced carbon-carbon with insulation and will cover one-third of the surface area.

At a thickness of 2.2 cm, the Thermal Analysis program showed that the shield itself did not protect the inside from overheating. A thin layer of insulation between the shield and the vehicle would provide adequate protection. The 1987 kg, insulated, reusable, non-ablative, carbon-carbon shield would be able to withstand temperatures of above 3500 K, according to NASA[2]. This would be sufficient for the M.U.R.P.H.S. reentry based upon past ballistic reentries[2].

The reusability of the shield is important. With the high mission frequency and the likelihood of many years of missions, a reusable shield becomes an economic necessity. Replacing shields after every mission would be costly and wasteful.

After the reentry is completed, the M.U.R.P.H.S. module would be further decelerated by a parachute released from the top of the vehicle. For this job, a remote, electronically-guided parafoil will be employed. This parafoil has many distinct advantages. First of all, such a parafoil has excellent vertical deceleration properties to assure a soft, safe landing of the module. Second, guided parafoils have a special asset in that they can be remotely piloted from the ground with pinpoint accuracy. The parafoil could cover over 100 km horizontally if necessary. This is a tremendous advantage due to the questionable accuracy of ballistic reentry when attempting a ground landing at a specific target such as an Air Force base. Parafoils such as this are considered by <u>Design News</u>[3] to be capable of large loads, such

as a reentry vehicle, and fully available by the 1990's.

Once the parafoil was fully deployed, the module would reach a vertical terminal velocity. The terminal velocity is related to the parafoil's effective area. An analysis is provided by Hoerner[1] and is represented by the following equation to estimate terminal velocity in English units.

$$U_t = 29(W/(C_D A_{eff}))^{1/2}$$

W is the load, A_{eff} is the effective area, and C_d is the drag coefficient which is about 2.4 for a parafoil of this size and type[1]. The problem of sizing the parachute is addressed graphically in figure RRS2.

From figure RRS2, a parachute of approximately 1300 m^2 would bring the module to a terminal velocity of about 10 m/s. This point on the graph was chosen for two reasons. First, it is a good trade point to minimize area and velocity. Second, from analysis in the Journal of British Interplanetary Science[4], a module of about the same structure as the M.U.R.P.H.S. crew could survive a complete retro-rocket failure. The crew would stand a good chance of survival during an impact at the terminal velocity . A load of 20,000 kg was used in the analysis. This very high estimate of down mass was used to allow for any possible overloading.

After the parafoil has brought the module to its terminal velocity, drogue parachutes help bring any horizontal velocity to near zero. The parachutes would then be released and retro-rockets would bring the module to a soft ground

Parachute Area vs. Terminal Velocity

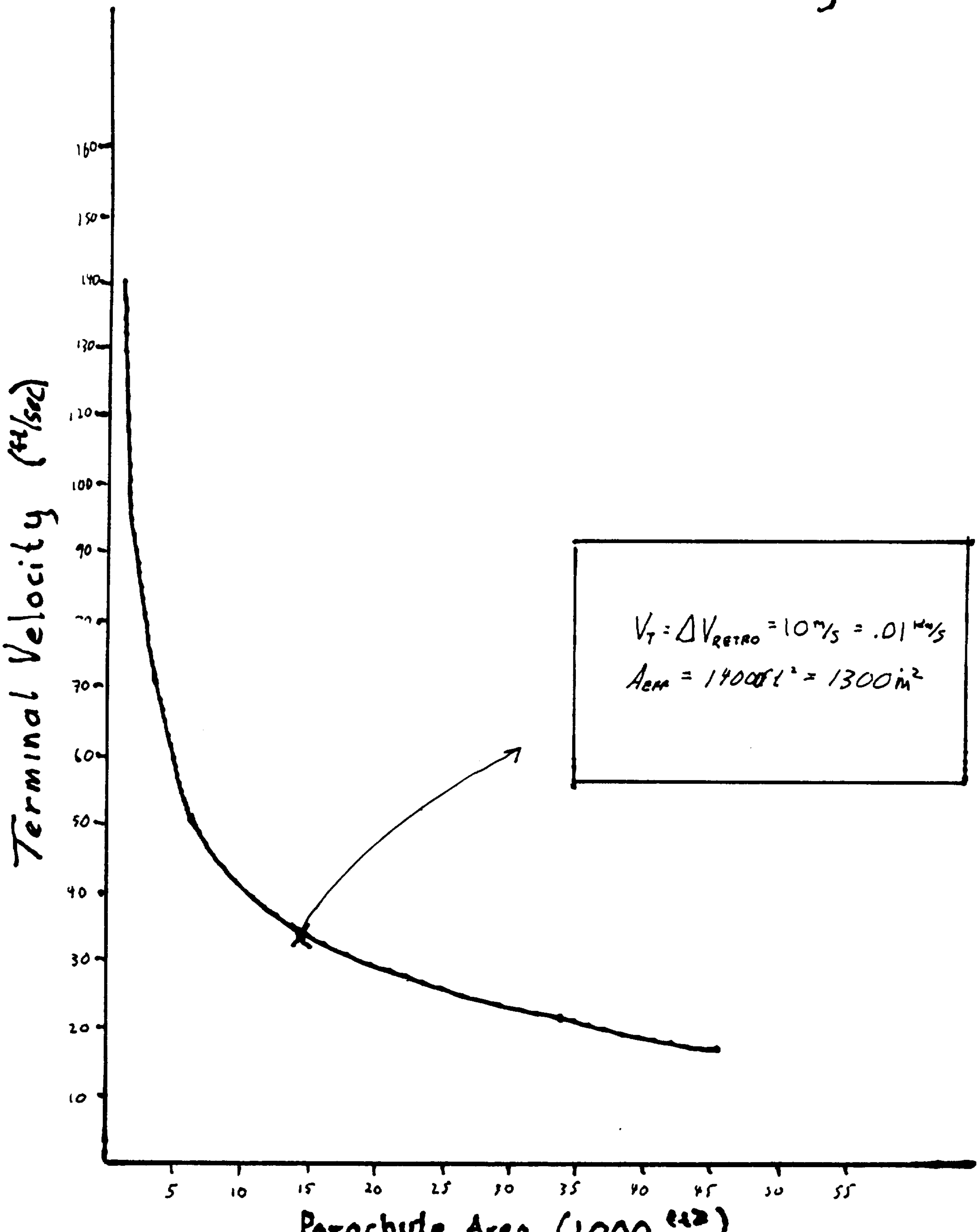


Figure RRS2.

landing on its retractable landing gear. The retro-rocket system would require a delta-V of .01 kg/sec. The thrust would be provided by four retro-rockets, from the power and propulsion subsystem, aimed off of the axis for stability. The thrust required is related to the parachute release height and the retro burn time. A trade is provided in figure RRS3. Higher thrust leads to higher g-forces but higher release height can cause stability problems.

At a height of 18.5 m the parachute would be released and the retro-rockets would be fired. The retro-rockets will provide 1.25 g's of thrust acceleration or about 211925 N of thrust for 3.75 seconds. The trust will be provided by the four retro-rockets and the module will be brought to zero velocity as it touches down on its landing gear.

The M.U.R.P.H.S. recovery system has many advantages over other systems. First, a landing on land at predetermined base allows an excellent and cost-effective ground support. Permanent landing sites could be developed to give a maximum of ground support. Water recovery, on the other hand, requires expensive naval assistance for every mission. While land recovery might be expensive at first, with a high mission frequency it would be better in the long run. A second choice for a land recovery would be a lifting body. Spending billions for what would essentially be another Space Shuttle is ridiculous; more shuttles could be built at a lower cost than designing a new lifting-body vehicle. The parachute/retro-rocket system would provide a

Parachute Release Height vs. Burn Time and Thrust Acceleration

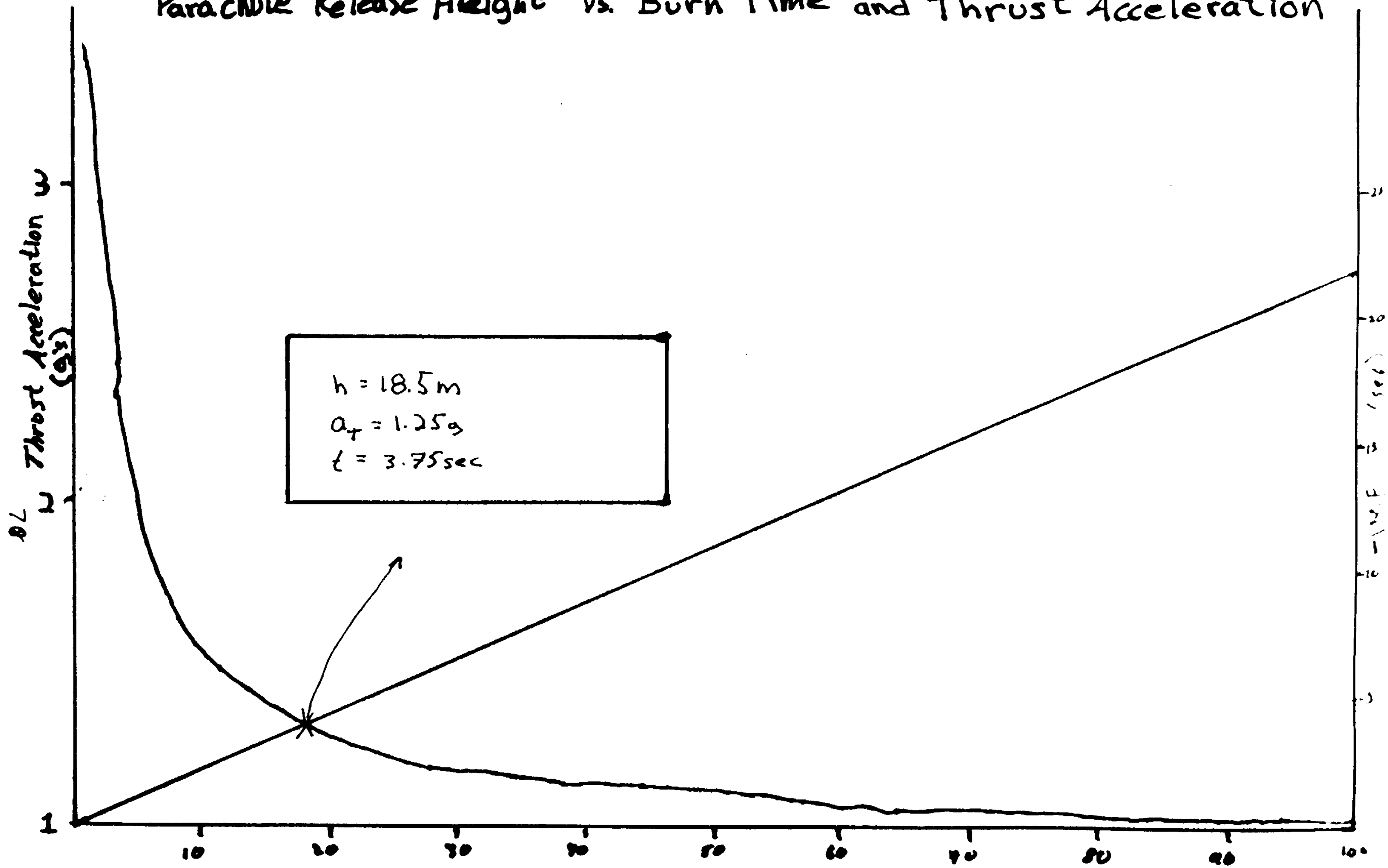


Figure RRS3.

safe, accurate landing with a total time on the order of eight hours--which is well within acceptable limits.

References

[1] Hoerner, Sighard F., Fluid Dynamic Drag, Hoerner Fluid Dynamics, 1965.

[2] NASA Facts, "Space Shuttle Thermal Protection," pp.48-52

[3] Design News, "Unmanned Parafoils Soft-lands Payload," March 3, 1986, pp.22-24.

[4] Journal of British Interplanetary Science, "Ballistic Reentry Vehicle Touchdown," November 1986, PP.36-41.

Command and Data Control System for MURPHS

The command and data control (CDS) system is responsible for the command of every system on the module. It also must provide communication between the various subsystem, and between the module and ground command centers. The discussion of this subsystem will begin with an analysis of the basic requirements of command and data control, followed by a discussion of the systems components, and finally an overview of modifications and problems that will need to be researched further.

I. Discussion of the Command and Data Control Basic Requirements

The requirements of the CDS system can be outlined as the following: collect telemetry from the various on-board systems, transmit telemetry to ground stations, relay and send commands to the subsystems, control power switching, and support crew interfacesand avionics. In addition this systems must control an auto-docking function for the module to dock with the space station or other orbiting platform.

A. Data Collection from On-board Systems

The satellite communication system is essential to the mission success. It includes various components of the other subsystems. The CDS system allows all of the subsystems to communicate to each other while it acts as a hub for the data transmission. These dialogs often circulate in closed loops and only reach the Earth as a summary of what happened long after the actual transmission; this is especially true when spacecraft status information is considered.

B. Transmission of Telemetry to Ground Stations

Information that must be conveyed between the ground and the module falls into three distinct categories: housekeeping or engineering data which tells about the health or status of the module, commands sent from earth or the space station, and navigation information. The engineering data normally consists of temperature, voltages, tasks being performed, or status of various systems. The navigation information will typically be the raw data from various sensors or gyros. In order to communicate this information to the ground, the spacecraft must contain a series of analog to digital convertors, data storage devices, data compressors, data display equipment, amplifiers, and an antenna.

This communication must be relatively error free; however, no finite amount of redundancy can guarantee perfect data transmission. To help insure error free transmission, several checking methods must be employed including parity bits, echo transmission, and redundancy. Luckily, most instrument readings usually vary slowly, so that erroneous data points will be easy to spot. The communication system will read data from other subsystems and instruments sequentially. This procedure is called commutation or multiplexing. This will produce a sequence to the data to be sent to the ground. A chart, Figure CDS-Two, showing a sample sequence follows:

CDS Figure Two: Data Transmission Sequence

	1	2	3	4	5
0	CDS Status	LSCS Status	PPS Status	AACS Status	Crew Communication
6	STRC-Temp	LSCS Status	PPS Status	AACS Status	Payload Status
11	STRC-Radiation	LSCS Status	PPS Status	AACS Status	Crew Status

In addition to the sequence shown here, several sub-multiplexers will be used to alter the data being transmitted from each subsystems during a particular sweep.

This telemetry format is very rigid with a certain number of bits assigned to each sequence box. This produces a lot of wasted bits that can be eliminated by intelligently changing the length assigned to each box on command and by using a data processor to automatically eliminate leading zeros from each word.

C. Relaying and Processing of Commands to the Related Subsystem

Commands for control of the module will fall into three classes: orbit control, attitude control, and spacecraft status control. The first two involve commands that are automatically relayed to the appropriate subsystem. The last class involves commands that must pass through the command system as they include standard housekeeping functions that generate stored commands that will carry out the desired function. The command subsystem also contains software capable of making necessary adjustment and decisions for the entire spacecraft and for itself including functioning of the antennas, power regulation, and and transmission rate and modulation. By definition the control subsystem carries out those decisions not assigned to the attitude and environmental control subsystems. Also, many of the other subsystems contain closed-loop control systems that do not involve the CDS system. See CDS Figures Three and Four.

II. System Components

Good communication systems can be measured by their reliability, cost, and data handling rates, but only when considered in a system context. This context leads to many design trade-offs. The measurements of a good system must be balanced against the system's weight, power, and compatibility requirements.

A. Computer/Processing Equipment

CDS FIGURE FOUR

76

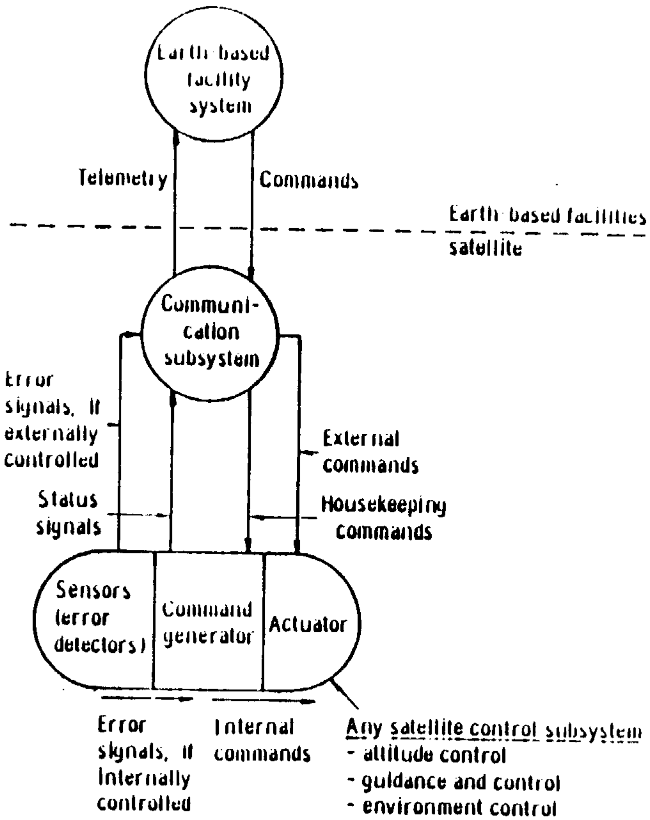


FIGURE 6-1.—Generalized block diagram of a satellite guidance-and-control subsystem. Internal and external control approaches are illustrated.

satellite attitude or battery voltage (fig. 6–1). This measurement is compared to the desired value of the parameter, which is specified by stored or communicated references. Any disparity above or below present limits causes control circuitry to direct a corrective command to the appropriate subsystem actuator. In closed-loop, or feedback, control, error signals are fed back until the performance deviation has been corrected. In open-loop control, commands are executed but there is no feedback; e.g., extension of the satellite solar-cell panels.

Specialized terms have evolved, particularly in orbit control. Table 6-1 defines these terms and establishes semantic boundaries between the three different control areas delineated above

CDS FIGURE THREE

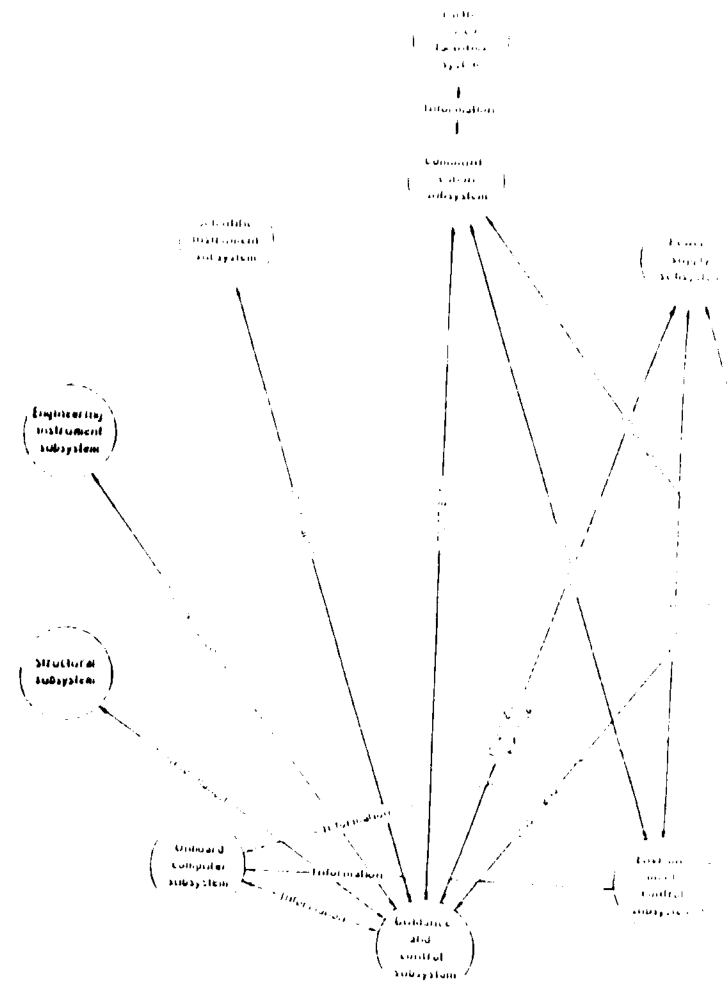


FIGURE 6-2. Interface diagram showing the more important ... ships between the three satellite-control subsystems and the ... of the spacecraft. The information interface includes the tra... and reception of error and navigational information as well a... mands.

nated or radio-triggered satellites (table 6-2). In all thr... the satellite first has to be visible in some part of the electr... netic spectrum to ground based instruments. Reflect... and self-contained optical and electromagnetic beacon ... satellite visible to cameras, the human eye, and radio ...

The first requirement of a data system is data storage and processing equipment. Most of the onboard processing requires little more computing power than a standard personal computer, and these can be used with minor modifications for the environment. But as this is a manned spacecraft operating in zero gravity and the extreme cold of outer space. Although the shell of the module will protect the crew and computer systems from the space environment, the systems must be designed to continue functioning should a breach in the spacecraft exterior occur. Two computer systems have already been developed to operate on scientific satellites that are not hardened to the environment. These are the SC-1 Spacecraft Computer developed by the Department of Space Science at Southwest Research Institute and NSSC-1, NASA's first Standard Spacecraft Computer selected in 1974. In addition to these the data processing facilities of the space shuttle orbiter need to be examined.

The SC-1 processor was developed for use aboard Spacelab and was intended to operate in the vacuum of space. Its general characteristics are found in CDS Figure 4. As the chart shows, the processor's memory is divided into three subsystems. It also provides an optional 2K bytes buffer. The self-scrubbing memory controller used with the DRAM manages all memory transactions as read/modify/write cycles so that corrected data and check bits will be constantly written back to memory. This helps the processor to identify and correct errors from subsystem inputs easily.

General Specifications for SC-1 Flight Computer

Configuration:
8086/8087/8089 tri-processor on local bus

Memory Capacity:
Onboard EPROM: 64 K (expandable to 128K)
Onboard DRAM: 128K (error correcting, single bit detect correct; multiple bit detect)
Onboard SRAM 2K bytes

I/O Capacity:
Parallel. 48 lines programmable (8255a), using two parallel interface adapters equipped to emulate an IBM-360 I/O channel handshake
DMA: Two 16 bit DMA ports, at 1 Mbps max t transfer rate
Serial: RS-232 port, controlled by USART for both standard asynchronous and synchronous communications

Interrupts:
Two 8-input priority interrupt controllers (15 hardware vectored interrupt lines available). Software configured for input priorities and mode.

Timer:
Two timers, each equipped with three 16-bit interval timers.

Power Consumption:
20 W

Weight:
9.38 lbs ()

CDS Figure Four: Specifications for SC-1 Flight Computer
NOTE: data taken from Gibson, p.212

In addition to the design considerations given to frequent error bits received from subsystems, the processor is also design to continue working in a spacecraft environment. It is mounted on a single 3/32" (0.24 cm) thick circuit board that is supported at 16 points to provide strength for the system during launch vibrations. An aluminum heat sink is attached to the circuit board and to the SC-1 case to conduct heat directly to the baseplate. This scheme allows the SC-1 to be operated in a vacuum where only conductive heat dissipation is possible. The computer will operate successfully with its base plate temperature between 85°C and -40°C as the plate is attached to the cabinet's structure.

The computer is designed to operate from a 28 V dc direct current. This current is passed through a dc/dc convertor which produces a 5 V dc current for the processor. Effort is also underway to harden the SC-1 processor to radiation to reduce its unshielded vulnerability. The computer uses a processor that is very popular in other applications and this allows the computer to support a variety of software written in languages like

Fortran, Pascal, Ada, C, and Unix. The advantage of this processor is that it will be sufficiently tested after having flown on numerous Spacelab missions.

The onboard data processing system on the space shuttle orbiter consists of four computers with another one reserved as a backup, known as GPCs (General-Purpose Computers). The software for the system is stored in two mass storage devices called MMUs and is retrieved when needed. Al of the computers are linked together using a 1 MHz data bus network. Most of the software on the vehicle is written in assembler or HAL/S, a high-order engineering language. Each of these computers weighs 120 pounds. These computers were developed in the late 1960's, and their combined computing power has been equalled since by a single desktop computer (Case Study: The Space Shuttle Primary Computer System, p. 899).

The only other standard processor used by the U.S. in space is the NSSC-1, NASA's first Standard Spacecraft Computer developed by IBM. It was first flown on a scientific shuttle payload in 1972. The CPU and most of the I/O logic are packaged on a 5x7 inch circuit board. The processor provides fully redundant applications with two processor modules and eight core memory modules of 64K. A HAL/S compiler for the NSSC-1 has been developed, but not been used yet (Case Study:..., p.902).

CDS Figure Five: Comparison of On-board Computer Systems

Criteria	SC-1	NSSC-1	Shuttle GPC
Weight	9.38 lb	unknown	120 lbs
Power	20 W ea	unknown	unknown
Hardened	Yes	Yes	No
Capability	High	Medium	Medium
Developed	ongoing	1974	1967
Language	many	HAL/S	HAL/S

It is obvious from an analysis of these three processing systems that the SC-1 computer is most applicable to the needs of MURPHS. The SC-1 processor is chosen as the primary processor for the MURPHS module because of its very low weight and power required. The system is hardened to the space environment to provide added safety if the environment system should fail on the module. It also allows for greater flexibility with its

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~~to the space environment to provide added safety if the environment system should fail on the module. It also allows for greater flexibility with its~~ modern configuration, components, and language/soft-ware support.

It is recommended that a separate SC-1 be devoted to attitude and articulation control, life support systems, and command control. An additional SC-1 should be left in reserve for redundancy. In addition to these four systems a desk top or personal computer should be used to provide for an intelligent crew interface to the modules computer system. The space shuttle utilizes a primitive keyboard and screen system that is very outdated. It is simple to replace this with a standard personal computer and provide more computing power to the module. A sample personal computer weighs 25 pounds and requires a 60 W power input.

B. Communication Systems

The space craft's communication system must be capable of providing almost continuous transmission and reception of information between the module and a ground station. This communication is accomplished by taking data from a memory buffer in the modules data processing system and sending it to the ground through an antenna. In between these two components are a number of smaller systems including modulators, digital/analog convertors, and amplifiers. The basic measurements of this system include reliability, power inputs and outputs, weight, cost, and compatibility.

There are essentially two methods of transmitting the data: microwave and laser technology. In addition there are two routes that can be used: direct to the groundbase and use of a data relay satellite. Fortunately, some research has been done on these options. If the system is designed to communicate directly to the ground, it must be designed with different parameters. Normally, the receiver antenna will be the monstrous Goldstone Antennas arrayed around the U.S. There is one system of data

different parameters. ~~Normally, the receiver antenna will be the monstrous Goldstone Antennas arrayed around the U.S. There is one system of data~~ relay satellites currently in orbit, these are the two Tracking and Data Relay Satellites (TDRSS) located in geosynchronous orbit developed for NASA by TRW.

The two categories of antennas are laser and microwave. There are three types of lasers whose technology currently meets our needs: FD Nd:YAG, HeNe, GaAs, and CO2. Some data concerning these lasers are shown in CDS Figure Six (Ponchak, and Spence).

CDS Figure Six: Characteristics of Laser Systems

Laser	Wavelength μm	Ave Power Output (mW)	Transmitter Efficiency	Lifetime in hours
FD Nd:YAG	0.53	100 to 500	0.5 to 2%	50,000
HeNe	0.63	2 to 5	1%	75,000
GaAs	0.83	20 to 10	5 to 10%	50,000
CO2	10.6	1000 to 2000	10 to 15%	5,000 to 10,000

The FD Nd:YAG diode pumped laser and the GaAs semiconductor laser were chosen to be the most suited for ~~the~~ our purposes, primarily because these approaches are amenable to simple and efficient direct-detection techniques. the required receiver is essentially a "photon bucket" and the phase of the received signal is unimportant. Also the amount of support equipment required for the CO_2 laser and the low output power of the HeNe laser remove them from consideration.

Some basic conclusions can be derived from a study of the various types of antenna classes. First, laser systems using a smaller diameter optics generally weigh less. Lasers are essentially insensitive to distance, while microwave transmission is highly sensitive to distance in its power and weight requirements. Therefore, for microwaves it is advantageous to use a larger diameter antenna to allow a lower output power requirement from the TWTA tubes that provide the signal. CDS Figure Seven allows us to compare the different systems if they operate under the same requirements

~~the TWTA tubes that provide the signal. CDS Figure Seven allows us to compare the different systems if they operate under the same requirements~~ of an intersatellite link at 1Gbps transmission.

CDS Figure Seven: Characteristics of Various Communication Systems

System	Diameter	Required EIRP (dBW) 10^{-7} BER	Transmit Power	Prime Power W	Weight kg
Laser (FD Nd:YAG)	6 inch	94 to 105	4 to 55 mW	80 to 95	47 to 50
	12 inch	88 to 99	0.3 to 3.5 mW	85 to 86	68
	24 inch	82 to 93	0.02 to 0.22 mW	90	180
Laser (GaAs)	6 inch	94 to 105	11 to 136 mW	80 to 90	47 to 50
	12 inch	88 to 99	0.7 to 8.5 mW	85 to 86	68
	24 inch	82 to 93	0.04 to 0.5 mW	90	180
23 GHz	3 ft	66 to 77	117 to 1476 W	300 to 3700	76 to 660
	4 ft	63 to 74	33 to 416 W	90 to 1000	35 to 220
	5 ft	61 to 72	13 to 168 W	40 to 400	28 to 100
32 GHz	3 ft	66 to 77	61 to 763 W	150 to 1900	48 to 350
	4 ft	63 to 74	17 to 215 W	42 to 540	27 to 120
	5 ft	67 to 72	7 to 87 W	23 to 220	24 to 65
60 GHz	3 ft	66 to 77	17 to 217 W	50 to 560	25 to 120
	4 ft	63 to 74	5 to 61 W	18 to 160	20 to 50
	5 ft	61 to 72	7 to 25 W	10 to 70	22 to 35

From this data we can see that the power required by the laser is always larger than the required power for microwave transmission. This criteria alone, not to mention others such as complexity and reliability, allow us to select microwave transmission as the preferred method. Once this has been done, we can look at the signal path. Traditional satellite communication has used either S-band (1.55 to 5.2 GHz), x-band (5.2 to 10.9 GHz), and Ku-band transmission (13.75 to 15.25 GHz). Most systems use the S-band frequency for voice and range data that can use the lower data rate, and the Ku-band with its higher data rate for telemetry and payload data. Both ground base stations and the TDRSS system support C and Ku band transmission. A comparison of the systems required for direct transmission to the ground or through a intersatellite link (ISL) is shown in CDS Figure Eight.

~~transmission. A comparison of the systems required for direct transmission to the ground or through a intersatellite link (ISL) is shown in CDS Figure Eight.~~

CDS Figure Eight: Communications System Comparison

Parameter	C-Band		Ku-Band	
	Baseline	ISL	Baseline	ISL
Antenna Size, ft	5.4 by 2.3	6.6 by 5.0	1.8 by 0.7	2.2 by 1.6
HPA Power, W	5.0	2.0	20.0	7.2
Payload Power, W	376	150	1079	396
Payload Weight, lb	325	250	557	278
ISL weight margin, lb	----	75	----	279

The weight data included here is computed assuming that the system is composed the transponder system, the antenna subsystem, and the portion of the power subsystem dedicated to the transponders. From other data obtained form the same source, it can be shown that the 75 lb ISL weight margin for a C-band satellite corresponds to the estimated weight of a 60 GHz package consisting of a 5 ft antenna and a 25 W TWTA. This package will offer a 1 Gbps capacity with a 10-7 Bit error Rate for separations between the module and the ISL satellite as large as 140°. Also we can see that the 279 lb weight margin for a Ku-band ISL satellite will easily allow implementation of either a laser or microwave ISL package providing capacities in excess of 4 Gbps over all angular separations.

From this data it is obvious that tremendous savings in systems size, weight, and power can be made by sending the transmission through a intersatellite link instead of directly to the ground. It then naturally follows that the selection of the TDRSS satellite as a hub for communication is the preferred choice. We can also assume that the same antenna can be used for communication at both frequencies, which is indeed the standard system on most spacecraft. Given this design parameter, we can estimate the antenna to be 6.6 by 5.0 ft in size, require 396 W of power, and weigh 278 pounds. This system will provide us with a minimum of 20 W signal output and a minimum of 1 Gbps at C-band and 4 Gbps at Ku-band operation. Both of these are well above the needs of our system.

C. Crew Interfaces

Once the CDS system is capable of onboard data processing and relaying telemetry to ground stations, it must support certain crew interfaces. First and Foremost it must provide verbal communication between the crew of the module and the ground station. Second it must allow the crew to override or reset some of the software command functions. Finally it must also transmit to the ground certain information provided by the crew as necessary.

The most important function of a crew interface with a communication system is to provide secure and reliable verbal communication with the ground. This is a very low data rate communication, and can easily be accommodated by C-band transmission. The data retrieval must be incorporated into the processing sweep as needed when there are messages to be sent. This would be a simple software step incorporated into the data processing program.

Second, the crew must be able to access and alter many of the software systems of the module as needed. This is a vital step in the command loop. This input would allow the crew to change the destination of the module from the space station to an other orbiting platform. It would also allow the crew to query the status of the module at random intervals. This interface can easily be accomplished through the use of a standard personal computer in the crew module. This would provide a familiar interface and intelligent software and data handling mechanism for the crew to enter the modules command system with.

Finally there must be the ability for the crew to enter data as needed according to the mission plan. Most of this operation would be handled by the PC in the crew compartment. In addition to the data that can be inputted through the keyboard it would be advisable to have various medical information gatherers in the case of an injured astronaut being returned to Earth. Certain biomedical instruments should be placed near the crew seats to allow attachment to an injured crew member. These instruments would include a thermometer, pulse and blood pressure devices, and other instruments. These instruments could be read by an analog device and then converted to a digital signal. NASA at its Johnson Space Center has also been developing infrared transmitters and receivers for wireless optical cabin communication. This system uses gallium-aluminum-arsenide light-

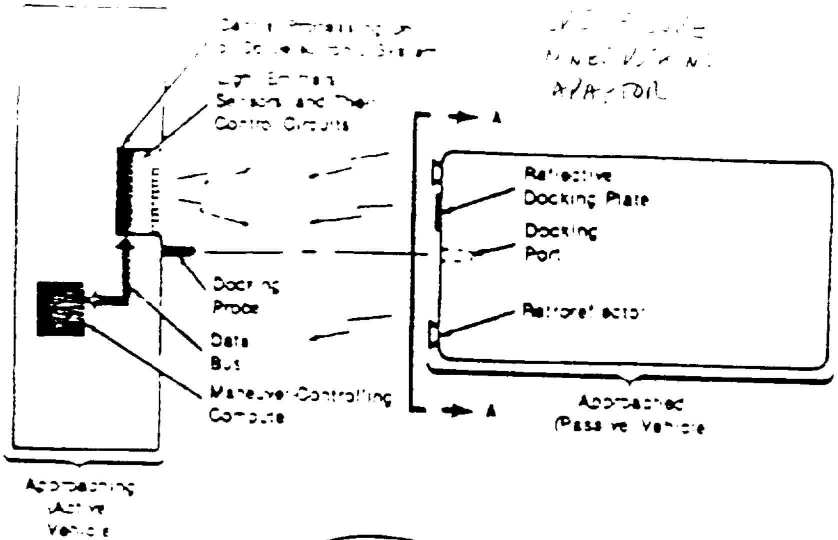
emitting diodes to send a signal through the cabin. Attached to each crew member would be a receiver the size of a cigarette package. This receiver would allow for wireless transmission of verbal communication and various vital signs to the modules data processor. Unfortunately, power and weight estimates of this system were not available and are presumed to be too large for our needs.

D. Docking Adaptor

One of the requirements of the system is that it be capable of autonomous docking with the space station. Although this is essentially an issue of attitude and articulation control systems, it is the commanded control of the various necessary maneuvers that require explanations and consideration at this time.

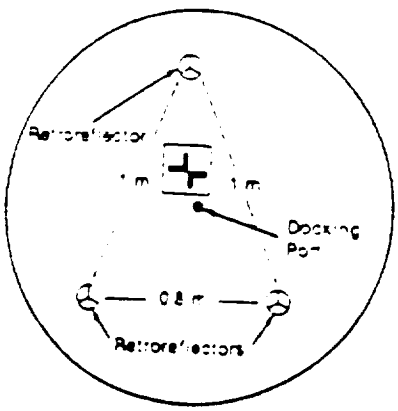
The Optoelectronic Docking System developed at Johnson Space Center automatically controls the approach and docking of an active vehicle with a passive vehicle ("Optoelectronic Docking System"). Our system communicates through the TDRSS system and can therefore receive rather accurate positioning data from this system. When the spacecraft are 30 km apart, the processor of the optoelectronic system will activate a pulse-laser ranging subsystem. Here a GaAs laser diode passes its light through a lens to create a fan-shaped beam of 18 to 20°. Through various sampling methods the direction to the target is determined to within 1° by 1° sector; the distance and closure rate are also determined. When the distance has closed to 30 m, the laser switches modes to a continuous wave, and tacks the signal reflected from three reflectors arranged in a triangle on the passive module. From these three return signals the system can compute the average target distance and its orientation relative to the active vehicles line of sight. Through more processing of the signal the direction to the target vehicle and its angle and roll rate can be found.

Once the separation distance has reached less than 3 m, a charged coupled-device television puls-ranging system takes command. The system processes the outline of the reflective docking plate in the television image to determine the target pitch and yaw rates. When the system moves to within one meter the television is too close to provide a clear image as the docking plate becomes larger than its field of view. Here though, alignment between four laser beams and the converging edges of a pattern in the docking plate becomes observable.



At 35 cm the docking probe enters the docking port and closes on a hard-docking indicator switch at 20 cm. This closing stops the system.

This system will require one additional processor to analyze the signals from the sensors; it will then really these analysis to the attitude and articulation processor to request the appropriate action. It will also require the other systems shown in CDS Figure Nine. 15



D. CDS System Components and Parameters Summary

After this lengthy discussion of the system components it is necessary to summarize the component selection and location conclusions.

The main component of the system is the interconnected computer/data processing network. This consists of five SC-1 processors: one each dedicated to attitude and articulation systems, life support systems, auto-docking mechanism, and command control. An additional SC-1 is placed ion the network to act as a backup system for safety purposes. In addition to these processors, there are two personal computers in the loop. One is for command control and system software, while the other provides a sophisticated crew interface. This system will provide more than adequate computing power at an affordable power and weight allowance.

This computer system is supported by a series of data measurement devices and analog to digital convertors that allow the processing network to monitor all onboard functions. There are also extensive data relay networks allowing the command system to relay commands to the various subsystems. This allows the command subsystem to regulate power consumption and subsystem actions through the use of various software running on the processing system.

Much of this processing is contained on a closed loop systems that does not require any input from a ground station to effect commands. However, some functions will require input from the ground. In addition much data must be transmitted between the module and the ground station to allow adequate monitoring of the onboard systems. Finally this interface is important to allow for manual overriding of the onboard functions. This interface is accomplished through the communications subpart of the command system. This system essentially consists of an antenna that allows the data relay. This antenna is located on the top portion of the module, even with the instrument compartment of the module. This placement allow the antenna to be protected from the heat of reentry, while still permitting it to track with a large degree of freedom. It also places it as close as possible to the data processing network to allow for easier transmission of data between the two. For further information on the exact placement, see the Structural Subsystem presentation of this report.

The final portion of the command subsystem is the docking adapter mechanism. This essentially consists of various sensors that allow the command system to determine the distance, attitude, velocity, and deflection between the MURPHS module and the space station during autodocking maneuvers.

CDS Figure Ten:Command and Data Control Subsystem Specifications

Component Name	Purpose	Size (m)	Mass (lbs)	Power (W)	Location (Compartment)
SC-1 Processor	AACS Control		4.26	20	Instrument Comp.
	LSCS Control		4.26	20	Instrument Comp.
	Auto-docking		4.26	20	Instrument Comp.
	CDS System		4.26	20	Instrument Comp.
	Backup System		4.26	20	Instrument Comp.
PC System	CDS processing	0.1x0.3x0.5	11.36	60	Instrument Comp.
	Crew Interface	0.1x0.3x0.5	11.36	60	Instrument Comp.
Multiplexer(5)	Control Data Input		11.36	--	Instrument Comp.
	Antenna Input		2.27	--	Instrument Comp.
Antenna	Telemetry Relay	2.0x1.5	126.36	396	Module Exterior
Docking Adapt	Auto-docking	1x1	--	--	Module Exterior
Totals			184.05	616	

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This figure lists all of the primary components of the CDS System. Their combined weight is 184. 05 kg (404.9 pounds) and their combined power requirement is 616 Watts. Since, these figures are based on estimates, and there are numerous other minor components of the system it is advisable to include an error margin to these totals. For this reason, the total weight of the CDS System should be estimated as 227.27 kg (500 pounds), while the power requirement should be 650 W.

III. Items to be Further Researched

This is a preliminary design and as such is subject to a lot of error and estimation. Before the system is finalized, there are certain areas that require more research and study.

Most importantly, the components of the system need to be tested for compatibility, and tested against the requirements to make sure they will work. Also, there must be other component options not considered in the body of this report that should be compared to the selected components. Also since the components in this report were hypothetical and not actually tested in a laboratory, the actual equipment needs to be analyzed. The exact weight, cost, power requirements, data rates, and computing capacity need to be measured to allow more accurate trade studies between various components to be undertaken.

The exact specifications for use of the TDRSS System needs to be figured into the component selection for the communication system. Compatibility and efficiency need to be measures.

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